V/STOL LIFT FAN COMMERCIAL SHORT-HAUL TRANSPORTS

Continuing Conceptual Design Study

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16. Abstract

A design study of commercial V/STOL transport airplanes for a 1985 operational time period has been made. The baseline mission considered was 400 nmi at a cruise speed of M = 0.75 and a 100-passenger payload with VTOL. Variations from the baseline included mission distance, payload, cruise speed, and propulsion system failure philosophy.

All designs used propulsion systems consisting of multiple gas generators driving remote tip turbine lift and lift/cruise fans. By considering the fan to be designed for "operational reliability," significant simplication of the airplane systems and reduction in airplane size and cost can be achieved.

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SUMMARY

V/STOL lift fan commercial transports for operation in 1985 were designed to determine technical and economic characteristics and to provide a consistent set of airplanes for operational studies.

A number of aircraft were designed for a baseline mission from which parameters of interest were varied. The airplane designed for the baseline mission carries 100 passengers, cruises at M = 0.75, and has a range of 400 nmi from a vertical takeoff and 800 nmi from a short takeoff. The propulsion system is made up of gas generators driving remote-tip-turbine lift and lift/cruise fans.

Variations from the baseline have involved mission length, cruise speed, payload, and fan failure philosophy. The influence on the airplane design of fan failure philosophy was most important.

The original designs achieved safety from fan failure by use of emergency jet nozzles which replace the nonoperating fan. These aircraft had a minimum of six fan/gas generators plus emergency nozzles. The concept of "operational reliability" for a fan extends to rotating parts the structural techniques and philosophy currently being used on wings and other primary structure. By designing with multiple load paths, use of wear and crack detection methods, as well as routine inspections, it has been possible to ensure that the load-carrying capacity of the primary structure will not be reduced below a safe operating level during a critical maneuver or between inspections. Applying this philosophy to fans, it is possible to drastically alter the V/STOL design.

The thrust/weight ratio required is set by the emergency condition. In figure 1 the thrust/weight ratio needed to achieve a thrust/weight ratio of 1.1 after a failure is shown as a

function of the number of fan/gas generators used and fan failure philosophy. The interesting point is that a design for safe fan failure with six engines needs the same thrust as an airplane with four "operationally reliable" fans.
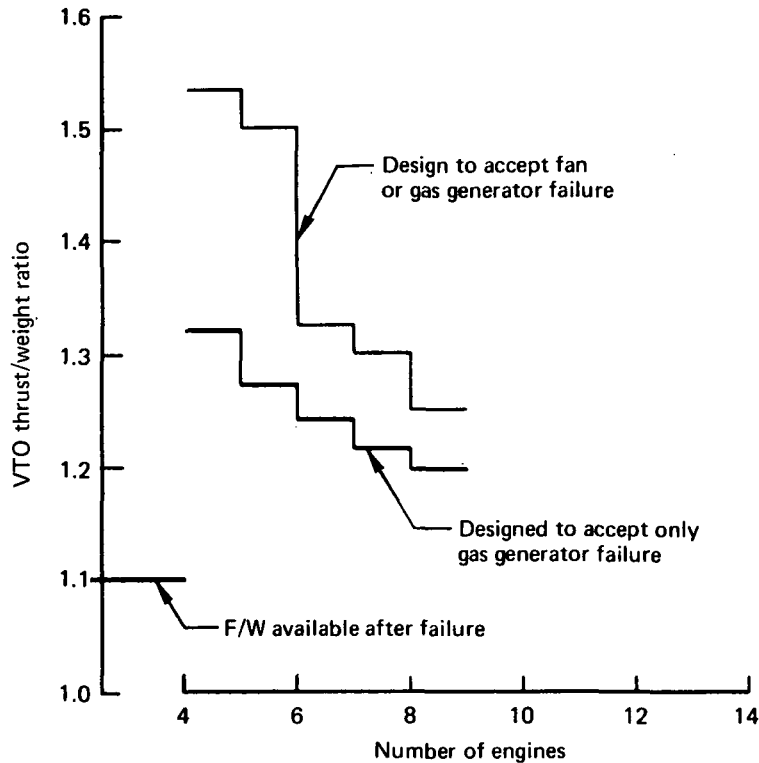
FIGURE 1.—THRUST/WEIGHT REQUIRED

The resulting six- and four-engine designs are shown in figure 2. The ability to design with four fans, at a reasonable installed thrust, led to various design simplifications. The folding forward lift fans are removed and the interconnect ducting is limited to the wings and carry-through structure. When both the four- and six-engine airplanes are designed with "operationally reliable" fans for the same mission, the four-fan airplane is slightly heavier than the six—about 4% at maximum VTO gross weight. This results from the design differences that cause the four-engine airplane to have shorter reaction control moment arms, lighter wing loading, and a higher cruise thrust loading. In spite of this, the manufacturing cost and DOC are about 8% lower than those of the six-engine airplane.

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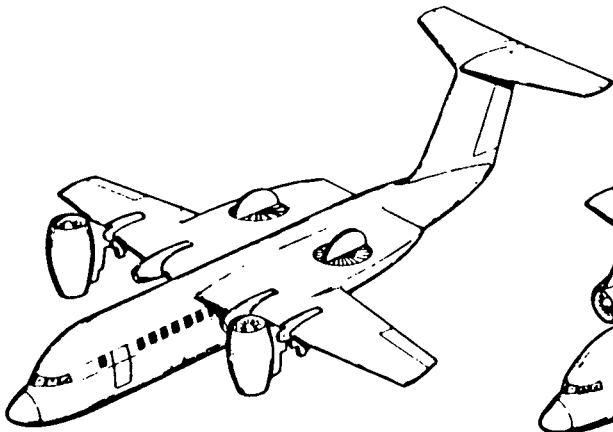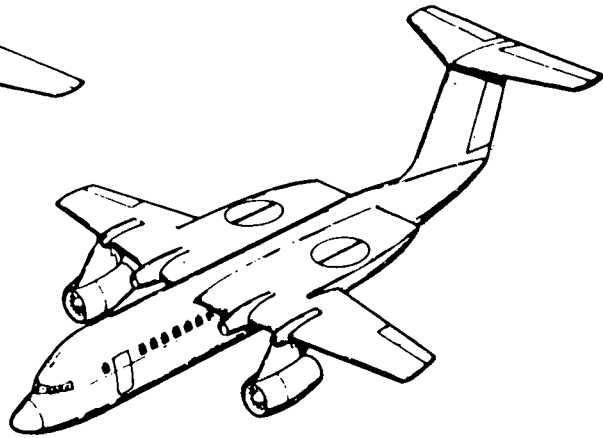
VTOL   CRUISE
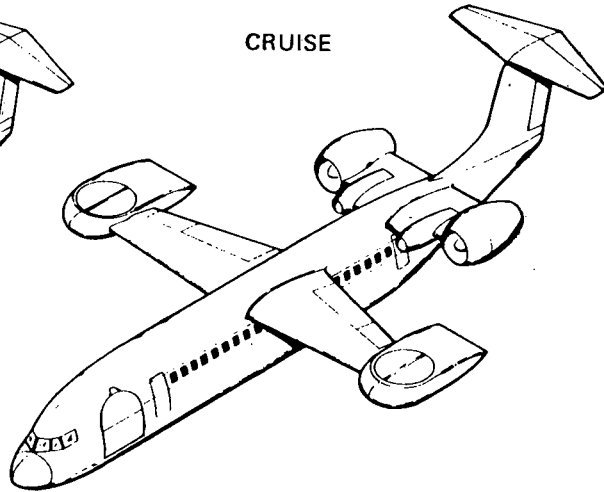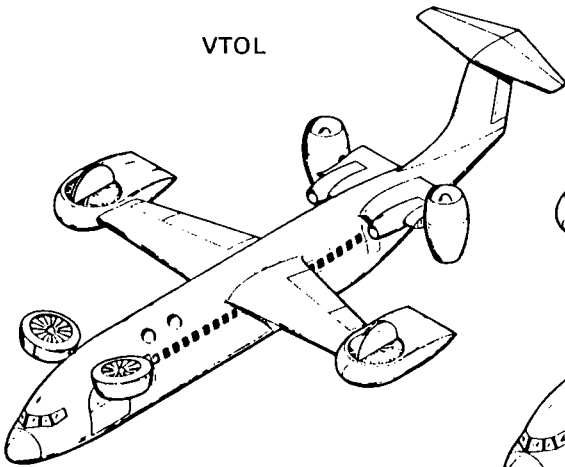
FIGURE 2.—FOUR- AND SIX-ENGINE AIRPLANES

The other design variations were made with the six-fan design as a baseline. To increase cruise speed from M = 0.75 to M = 0.85 requires sweeping the wing, refairing the wing-fuselage and fuselage-tail junctures, and refairing the flight cab. These changes can be achieved at a weight increase of about 8%.

Reducing the cruise speed to M = 0.65 can result in a small, about 1%, saving in mission fuel. This benefit is achieved by redesigning to optimize for the lower speed. The same result would be obtained by operating the baseline airplane at M = 0.65, without redesign.

Designing for 150 passengers instead of 100 increases the gross weight by 52%. The larger airplane is nearly a direct scaled version of the six-engine baseline.

A useful steep approach corridor is available. Maneuver and controlled descent can be accomplished at flightpath angles down to minus 24°.

Even from a noise standpoint, introduction of these airplanes into a community will have a salutary effect. The takeoff and landing 95 PNdB noise contour encloses an area more than 100 times smaller than that of conventional short-haul jet transports and is within the boundaries of 96% of the short-haul airports.

Initial cost of V/STOL· transports will be about twice that of conventional airplanes. This is almost entirely due to the cost of propulsion, which is about 15% of the cost on conventional aircraft but is about 50% of these airplanes.

Direct operating costs are shown in figure 3. The DOC as a function of range is shown for the six-engine baseline airplane and the four- and six-engine short-range airplanes with "operationally reliable" fans. A band representing CTOL transports is included for comparison. The V/STOL operating costs are 30% to 100% above current aircraft at a range of 200 to 300 nmi.
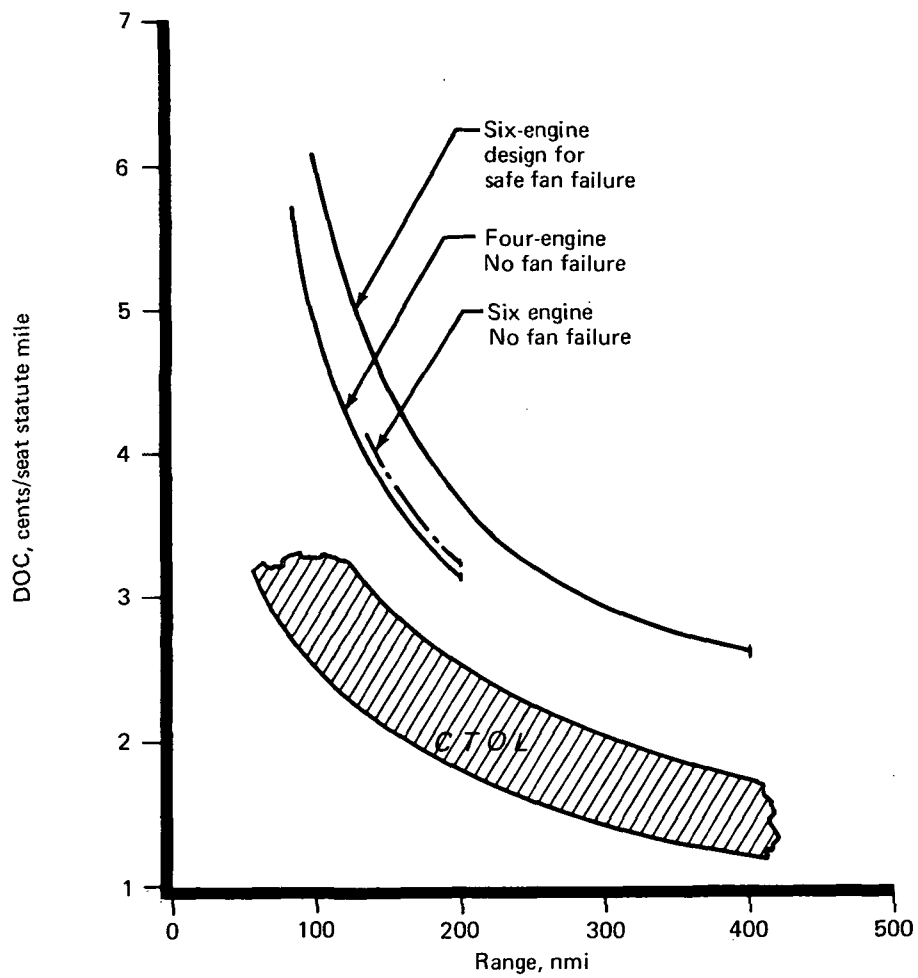
FIGURE 3.—DIRECT OPERATING COST

SYMBOLS

A_{des} design nozzle area, feet^2 (meters^2)

A_F fan nozzle area, feet^2 (meters^2)

a_n acceleration normal to the flightpath, feet/second^2 (meters/second^2)

a_p acceleration along the flightpath, feet/second^2 (meters/second^2)

A_{prim} primary nozzle area ratio, A_p/A_{des}

A_{sec} fan nozzle area ratio, A_F/A_{des}

b wing span, feet (meters)

BPR engine bypass ratio, W_e/W_p

C_D drag coefficient, D/qs

C_L lift coefficient, L/qs

cg center of gravity

$CTOL$ conventional takeoff and landing

D drag, pounds (newtons)

DOC direct operating cost, cents per available seat statute mile

F thrust, pounds (newtons)

F_{CL} climb thrust, pounds (newtons)

F_{CR} cruise thrust, pounds (newtons)

F_g gross thrust, pounds (newtons)

F_{MC} thrust at max control rating, pounds (newtons)

F_n net thrust, pounds (newtons)

F/W   thrust/weight ratio, pounds/pound (newtons/kilogram)

hp   horsepower (watts)

Ix, Iy, Iz   moments of inertia, slug feet^2 (kilogram meters^2)

I_{xz} cross product of inertia, slug feet^2 (kilogram meters^2)

L/D   lift drag ratio

L   lift, pounds (newtons)

L   characteristic length

$\mathcal{L}$ rolling moment, foot-pounds (newton-meters)

ℓ moment arm

LRC   long range cruise speed. The higher of two speeds at which 0.99 best nautical miles/pound is achieved.

M   Mach number

M   pitching moment, foot-pounds (newton-meters)

MAC   mean aerodynamic chord

N   number of engines

n   yawing moment

OEW   operating weight empty, pounds (kilograms)

PNL   perceived noise level

PNdB   perceived noise level, dB re 20 micronewtons/meter^2

q	dynamic pressure, pounds/foot^2 (newtons/meter^2)
R_C	compressor pressure ratio
R_F	fan pressure ratio
$R_{overall}$	fan + compressor pressure ratio
R/S	rate of sink, feet per minute (meters/second)
S	wing or reference area, feet^2 (meters^2)
SAS	stability augmentation system
SFC	specific fuel consumption, pounds/hour/pound (kilograms/second/newton)
STOL	short takeoff and landing
T	temperature, degrees
TAS	true air speed, knots (meters/second)
TIT	turbine inlet temperature, degrees
t/c	thickness/chord ratio
TOGW	takeoff gross weight, pounds (kilograms)
V	flight speed, knots (meters/second)
V_{AP}	approach speed, knots (meters/second)
V_{MCA}	minimum control speed in the air, knots (meters/second)
V_{MCG}	minimum control speed on the ground, knots (meters/second)
V_{min}	minimum flying speed, knots (meters/second)
V_R	rotation speed, knots (meters/second)

V/STOL	vertical/short takeoff and landing
VTO	vertical takeoff
VTOL	vertical takeoff and landing
VTOGW	vertical takeoff gross weight, pounds (kilograms)
W	weight, pounds (kilograms)
W_{AT}	total engine airflow, pounds/second (kilograms/second)
W_e	fan airflow, pounds/second (kilograms/second)
W_f	fuel flow, pounds/hour (kilograms/hour)
W_p	primary air flow, pounds/second (kilograms/second)
α	angle of attack, degrees (radians)
β	elevation angle between the noise source and a sideline listening point, degrees (radians)
γ	flightpath angle, degrees (radians)
θ	pitch angle, degrees (radians)
$\Lambda c/4$	sweep of the quarter chord line, degrees (radians)
¢/ASM	cents per available seat statute mile
σ	gross thrust vector angle relative to the horizontal body reference line: when thrust is horizontal and forward, $\sigma = 0°$; when thrust is vertical and up, $\sigma = 90°$ (fig. 37), degrees (radians)
ϕ	roll angle, degrees (radians)
ψ	Yaw angle, degrees (radians)

STUDY SCOPE AND PURPOSE

V/STOL lift fan commercial transports for 1985 were designed to determine technical and economic characteristics and to provide a consistent set of airplane designs for operational studies. Baseline conditions were set from which the parameters of interest were varied. The airplane designed for the baseline mission carries 100 passengers, cruises at M = 0.75 or at an equivalent airspeed of 350 kn (whichever is less), has a range of 400 nmi from a vertical takeoff or 800 nmi from a short takeoff. The propulsion system consists of gas generators driving remote-tip-turbine lift and cruise fans. The fans and gas generators are interconnected to permit power transfer for control and to provide balance in the event of an engine or fan failure.

The variations from the baseline involved mission length, cruise speed, number of passengers, and fan failure philosophy. The work was accomplished in steps, which permitted a limited coverage of the variables, rather than attempting to study a complete matrix.

DESIGN VARIATIONS

Specific performance and philosophical changes were made from the baseline. The first variation from the baseline mission was to increase the cruise Mach number from 0.75 to 0.85. Holding the cruise at M = 0.75, the effect of increasing the payload from 100 to 150 passengers was next examined. Then, at 100 passengers and M - 0.75, the design was modified by the assumption that the remote fans would not fail in flight. The effect on this airplane of reducing the range to 200 nmi and then reducing the cruise speed to M = 0.65 was also examined. This study process is represented schematically in figure 4.

The airplanes designed to tolerate a fan failure had at least six fans and gas generators as well as an emergency jet nozzle system. The airplanes that have operationally reliable fans, fans that will not fail in flight, were designed with six fans and gas generators as well as four fans and gas generators.

DESIGN GUIDELINES AND TECHNOLOGY

Design guidelines and technology levels were established by the Ames Research Center of NASA and The Boeing Company for the study.

Baseline
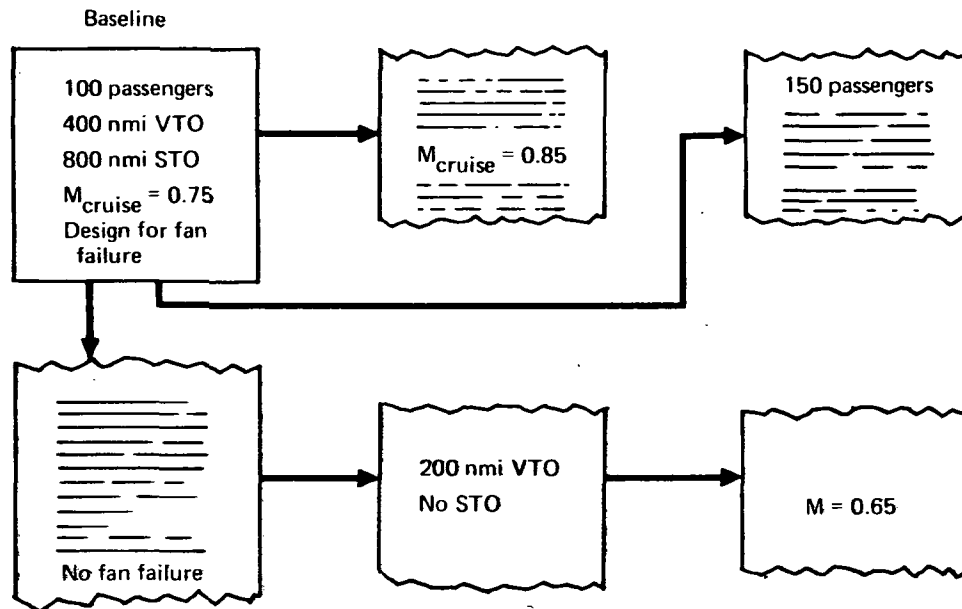
FIGURE 4.—STUDY TASK SCHEMATIC

Design Guidelines

The design guidelines established the mission calculation rules, the hovering thrust, and moment requirements and provided the basis for estimation of initial investment and direct operating costs. Pertinent extracts from the guidelines are presented in appendix A.

The mission requirements represent conventional practice modified to account for the V/STOL capability. The most significant mission change is in reserves. These have been reduced.

The requirement to continue flight after any single failure had a major influence on the installed thrust. In general it was specified that, during hovering after a failure, with the airplane

trimmed, the thrust exceed the weight by 5% while a specified level of control was applied about all three axes. That is, the thrust/weight ratio after a failure with emergency power is

$$\frac{F}{W} = 1.05 + \Delta\frac{F}{W}\,\text{trim} + \Delta\frac{F}{W}\,\text{applied control} \tag{1}$$

A similar relationship established the requirement with all engines operating. As would be expected, the failure case is usually critical, and the total installed thrust is that required to meet the hovering failure criteria.

Technology—State of the Art

The technology levels expected in 1985 were used in designing these aircraft. Estimations were made in the areas of structures and weight, propulsion, aerodynamics, and noise.

Structures and Weight Technology

The structural and weight technology level for 1985 is the same as that used in references 1 and 2, a forerunner to this study. The application of advanced structural materials resulted in a 16% reduction in structural weight from current practice. This level represents the best that can be expected for operation by 1985 in Boeing's opinion and is used throughout the study. At the request of NASA-ARC, the effect on the airplane weight of a 25% reduction was also considered for the baseline airplane. The application of the advanced technology to the airplane was made to provide the most effective use of the advanced materials. The application to these configurations is shown in figure 5.

Graphite-epoxy honeycomb is used for the wing, fuselage, and empennage primary structure. This led to a 26% weight saving for the wing, 19% for the empennage, and 17% for the fuselage. Graphite-epoxy is used for the non-temperature-critical area of the nacelles with the acoustic treatment integrated into the structure. This yields an 11% weight saving over a conventional nacelle. Conventional aluminum skin-stringer construction is used for the temperature-critical areas of the nacelle. The overall structural weight increment of 16% resulted from this application of the advanced materials.

Weight reductions of subsystems and fixed equipment for advanced technology are included in the analysis. Improvements are expected in flight controls, electronics, furnishings, secondary power systems, and standard and operational items of useful load. The effect of technology level on the airplane weight is illustrated in figure 6.

12

Model 984-139

Weight
improvement, %

▬		8
▒		11
▢		17
▨		19
▓		26

Structural
technology
level

Total structure   16
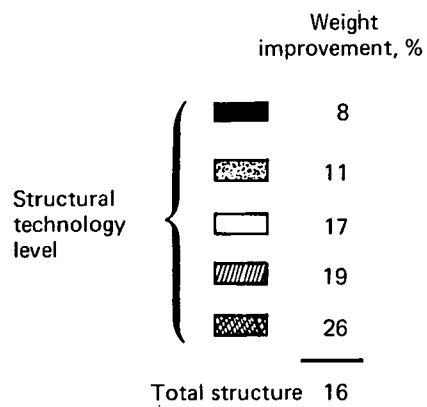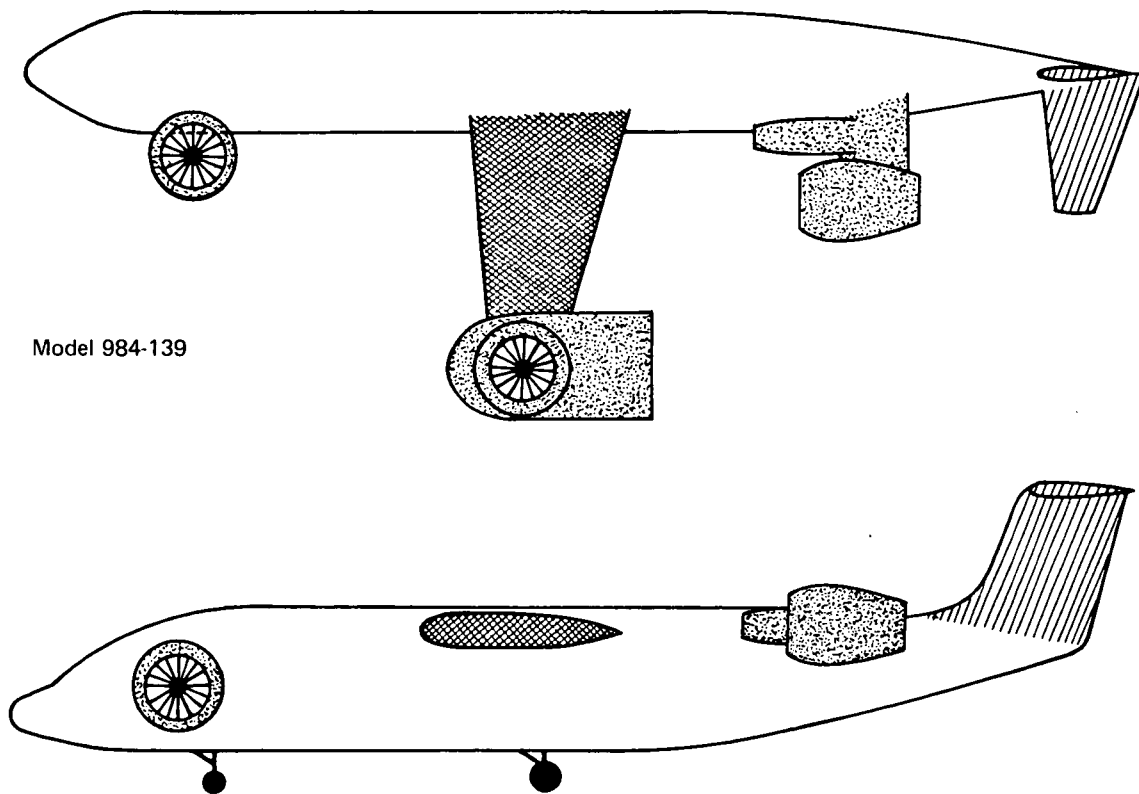
FIGURE 5.—ADVANCED TECHNOLOGY DISTRIBUTION (1985 OPERATION)

13

Model 984-139.
GE remote lift fan engines
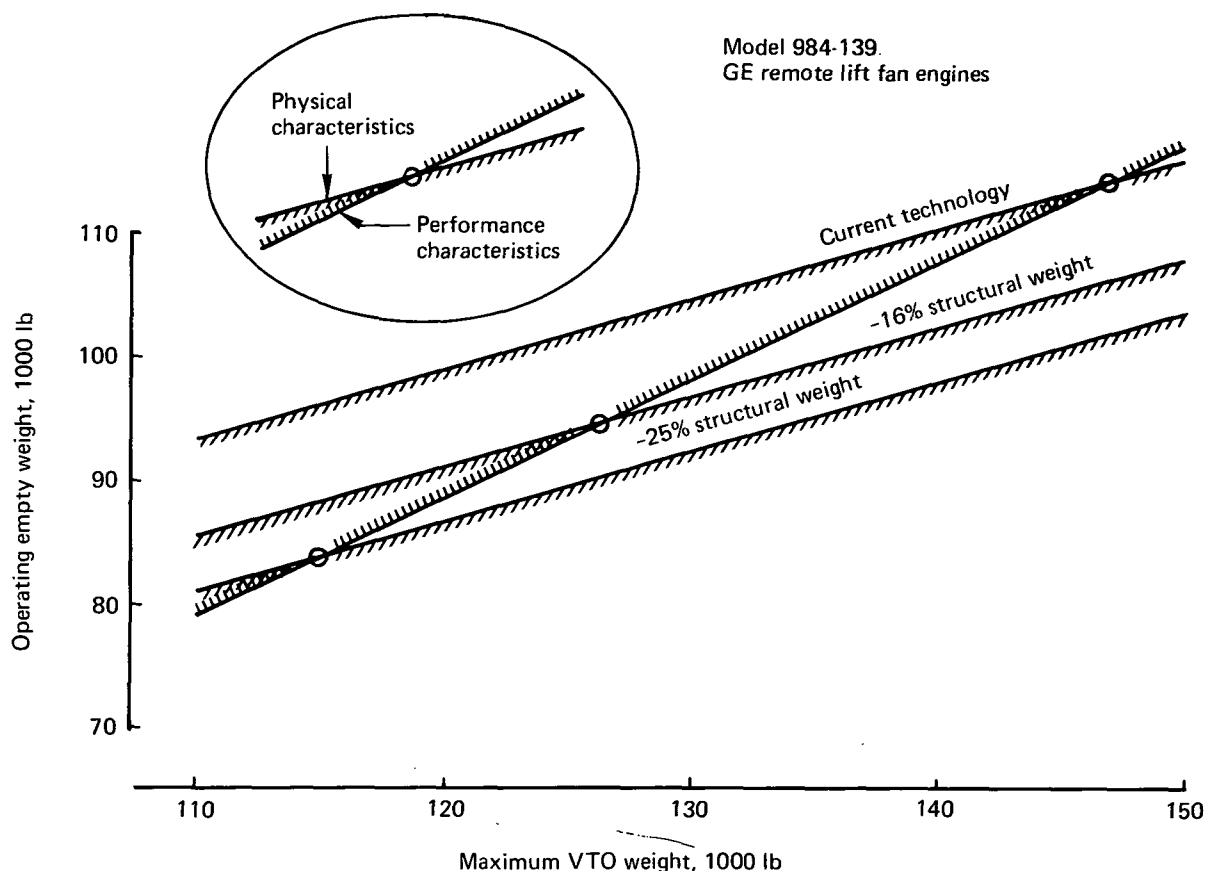
FIGURE 6.—ADVANCED TECHNOLOGY WEIGHT COMPARISON

The physical characteristics line indicates, for each level of technology, the relationship between operating empty weight and maximum VTO weight that must hold for the airplane to be feasible. The line indicates the minimum weights possible. Actual physical airplanes can only exist on or above the line. The performance characteristic line is the maximum weight relationship that will permit execution of the design mission; only airplanes on and below the line can achieve the required performance. The intersection of these lines defines the lightest airplane that can both be built and meet the mission.

The baseline 1985 operational airplane has an operating empty weight of 94 750 lb and a VTO gross weight of 126 300 lb. With current techniques these weights would be: OEW = 114 000 lb and VTO gross weight = 147 000 lb. The 25% reduction in structural weight results in OEW of 83 500 lb and a VTO gross weight of 115 000 lb.

Aerodynamic Technology

The airplanes are designed to a maximum wing-loading of 150 lb/ft^2. This is estimated as an upper bound at which a buffet-free maneuver margin is available at cruise speed and altitude. The wing and empennage airfoil sections are representative of the supercritical technology expected by 1985. With a straight wing of relatively thick section, the anticipated airplane aerodynamic efficiency is shown in figure 7.
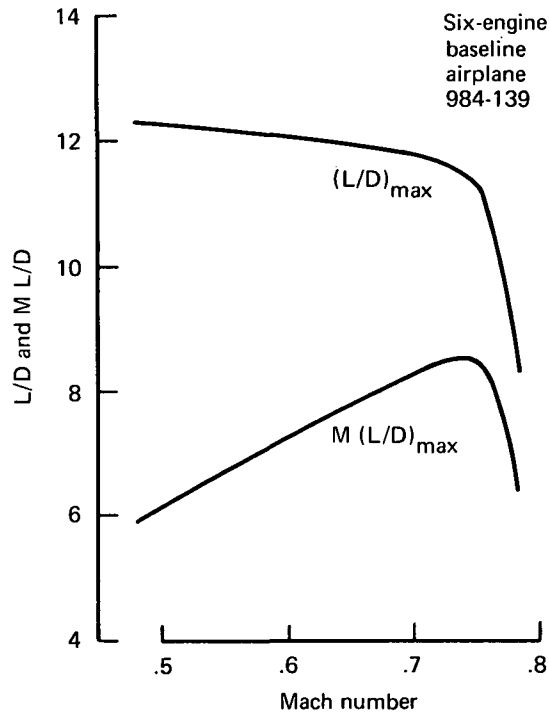
FIGURE 7.—AERODYNAMIC EFFICIENCY

The low-speed aerodynamic systems are current state of the art. It is anticipated that a limited-capability high-lift system will be required to provide adequate stall margins during terminal area flight and transition.

Propulsion Technology

The propulsion systems for this study consist of gas generators driving remote-tip-turbine lift and cruise fans. Performance developed for a family of fans with design pressure ratios of 1.25 to 1.35 was based on General Electric technology for 1985. The size, weight, and scaling rules are from references 3 and 4. A lift fan cross section from reference 3 is shown in figure 8. The study airplanes all have fans with a design pressure ratio of 1.35.
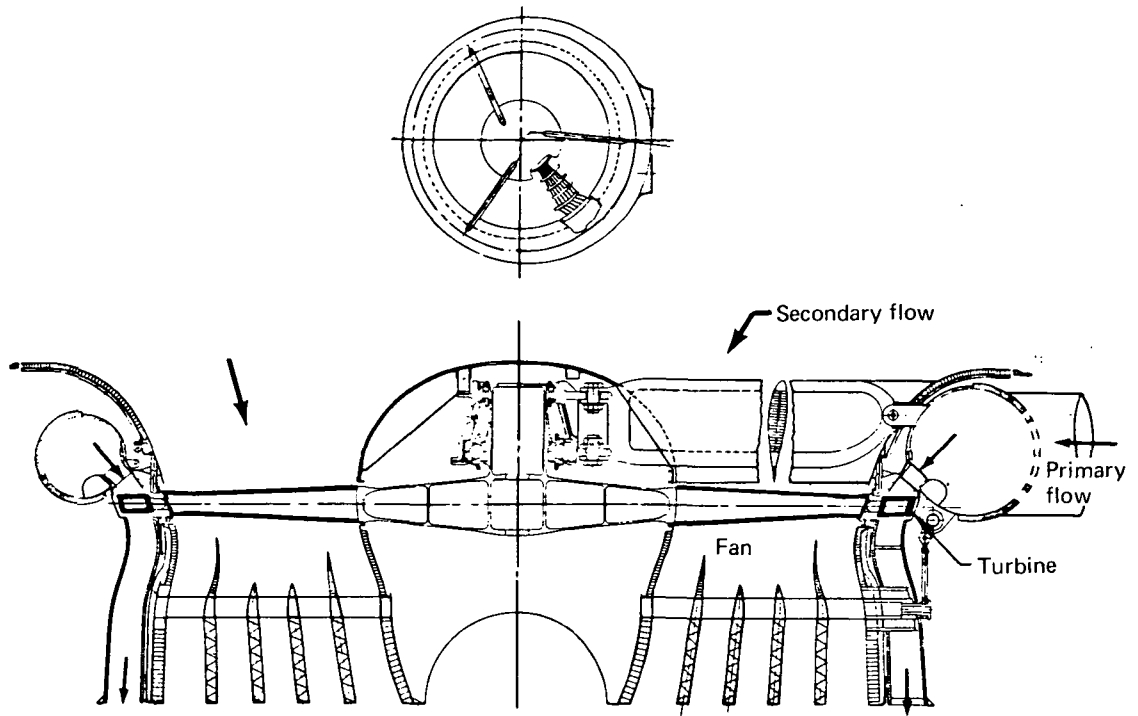
FIGURE 8.—1.25 PRESSURE RATIO LIFT UNIT CROSS SECTION

Weight and size are scaled from the basic reference thrust to other thrust levels by the static thrust ratio to the appropriate power.

$$W = W_{ref}\left[\frac{F_{MC}}{F_{MC_{ref}}}\right]^{1.25} \qquad (2)$$

$$L = L_{ref}\left[\frac{F_{MC}}{F_{MC_{ref}}}\right]^{0.5} \qquad (3)$$

The fans are designed at pressure ratios of 1.25, 1.30, and 1.35 at maximum control thrust. The static thrust is flat rated to ISA + 31° at sea level. The gas generator has a compressor pressure ratio of 20 and a turbine entry temperature of 2755° R. This gives a duct and scroll temperature of 2060° R. This is the maximum allowable gas temperature for the duct and turbine scroll and is a transient occurring at the maximum control thrust rating (F_{MC}). An emergency rating 4% greater than this is assumed to be available. It will be necessary to inspect the duct scroll and fan turbines

after use. The duct and scroll temperature for maximum climb is 1930° R and for maximum cruise is 1860° R. These temperature limits have a limiting effect on the engine performance in terms of the available fan turbine power.

For each design condition, the bypass ratio was determined that would give equal primary and fan nozzle total pressures at the maximum control thrust. This condition provided for good mixing of the two streams and reasonable pressure in the primary nozzle at part power. The bypass ratio as a function of fan pressure ratio is shown in figure 9.
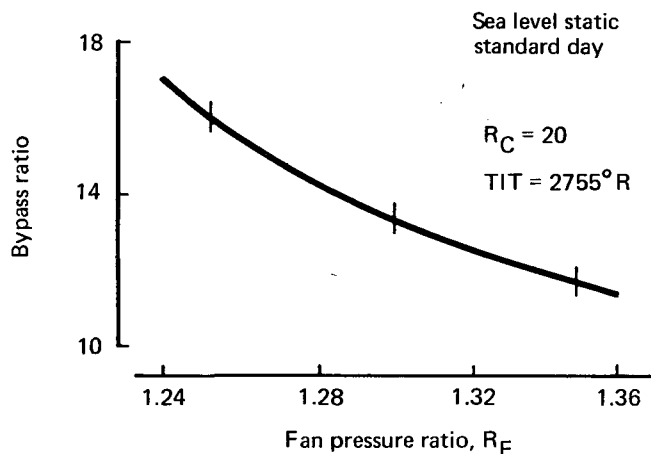
FIGURE 9.—FAN DESIGN POINT

The cruise and climb performance of these fans is attained by use of a two-position fan nozzle. The selection of the cruise nozzle area was made at M = 0.75 at 36 089 ft. The areas used are shown in figure 10. The primary nozzle area does not vary.
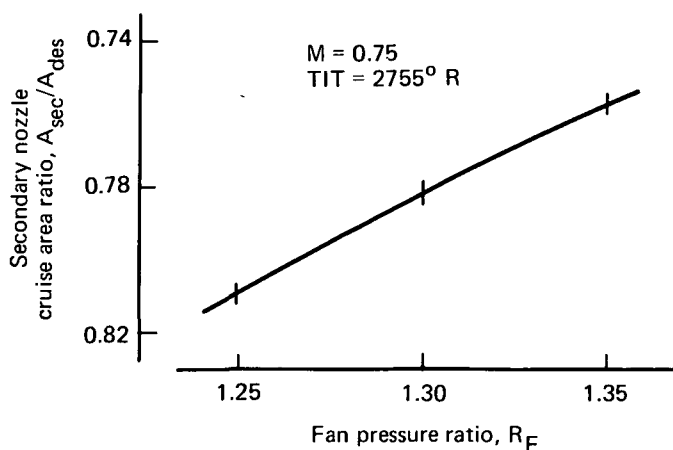
FIGURE 10.—CRUISE NOZZLE AREA RATIO—FAN

An attempt was made to improve the cruise performance by adding heat at the fan turbine. This implies that, for a cruise installation, the scroll could be modified to permit a straight-through flow so that the scroll walls could be cooled and the scroll temperature limit would not apply. The maximum temperature would then be set by the turbine blades. The location of the turbine at the end of the fan blades precludes turbine blade cooling. A maximum temperature of $2260°$ R was assumed possible without cooling. The effect of this heat addition on thrust and SFC at M = 0.75 at 36 089 ft is shown in figure 11. At the turbine temperature limit, the net thrust increases 4.8% and the SFC goes up 8.7%. The small thrust advantage, the relatively large SFC penalty, the complex scroll modification, and the addition of auxiliary burners and fuel control all added up to a large penalty rather than a gain. The auxiliary burner concept was dropped.

The complete performance, size, and weight information on the reference fan/gas generator sets is presented in appendix B. The net thrust lapse rate, which is characteristic of the cycle, is presented in figure 12 as a function of Mach number and altitude for the design fan pressure ratios.

Noise Technology

Noise technology for 1985 assumes the development of suppression techniques combined with good design that will substantially reduce the noise output of a fan/gas generator combination. A unit with a design fan pressure ratio of 1.25 and capable of producing 20 000 lb of thrust will have a noise signature at 500 ft of 91.2 PNdB. Without suppression or advanced technology this would be 111.5 PNdB. This noise base is from General Electric. It is the noise generated at the noise rating thrust level used during takeoff and landing maneuvers. In the General Electric study, this was arbitrarily set at 80% of the maximum control thrust; however, to generate the takeoff and landing noise contours, the actual thrust required (power setting) along a typical flightpath is used.

The use of design fan pressure ratios greater than 1.25 was accounted for as a noise increment on the base. This variation is shown in figure 13. The increment due to a change from pressure ratio 1.25 to 1.35 is 2.8 PNdB.

The effect of design pressure ratio on the power setting used along the flightpath is very slight. The difference in lapse rate from zero to 100 kn for fan pressure ratios 1.25 and 1.35 is shown in figure 14. This small difference will have a negligible effect on the power setting contribution to the noise as the design pressure ratio is varied.

M = 0.75 at 36 089 ft
Standard day

Max fan turbine blade temperature
(uncooled)
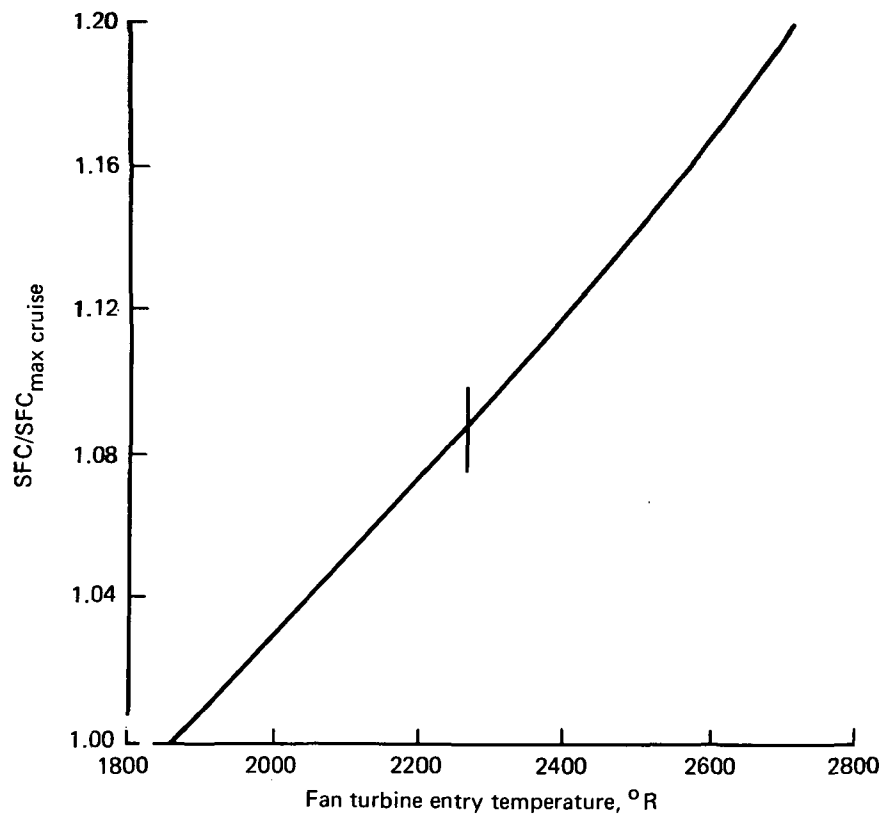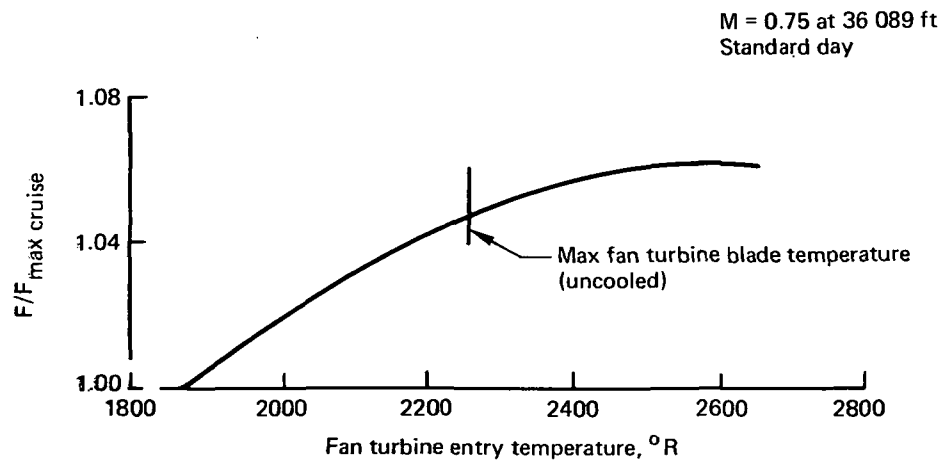
FIGURE 11.—AUXILIARY BURNER CRUISE PERFORMANCE

19
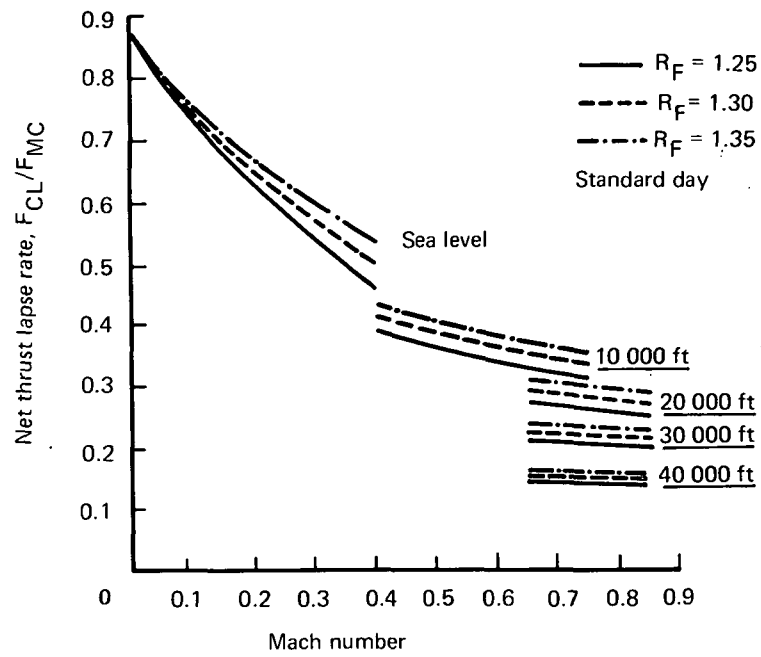
FIGURE 12.—NET CLIMB THRUST LAPSE RATE
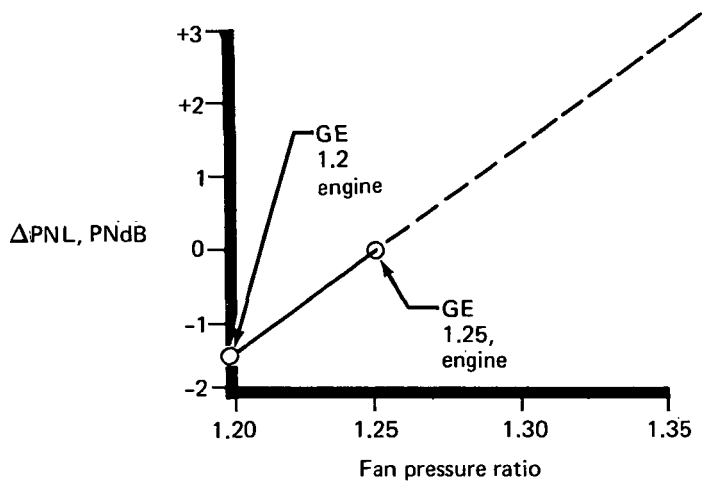
FIGURE 13.—NOISE INCREMENT OF FAN PRESSURE
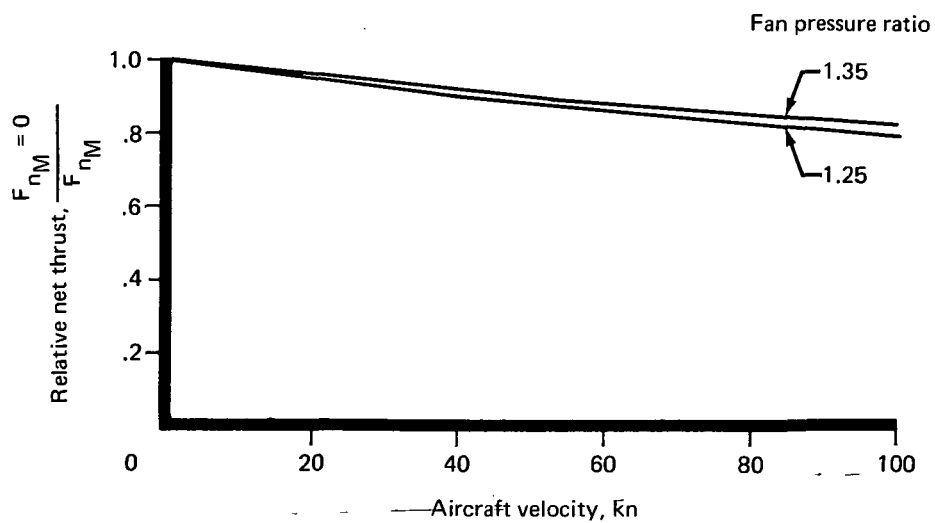
FIGURE 14.—TAKEOFF THRUST COMPARISON

DESIGN INTEGRATION

CONFIGURATION SELECTION

A number of airplane configurations were considered to meet the baseline criteria. The selected concept was then used, with the necessary modification, to fulfill various alternate requirements. In the preceding study (refs. 1 and 2), a remote lift/cruise fan airplane was configured with eight fans and gas generators, of which four were used in cruise.

To improve on that design, it was decided (1) to cruise on two engines and (2) to have only one kind of fan and gas generator for both lift and cruise. The engines used only for lift would be cowled differently than those used for both lift and cruise. The basic gas generators and fans would be identical.

The initial configuration selection considered number of engines and arrangement. The installed thrust required resulted from two different flight conditions: vertical takeoff and cruise. The vertical takeoff thrust was set by the fan failure condition.

The gas interconnect system, with emergency nozzles, combining two fan/gas generators is shown in figure 15. If a gas generator fails, the isolation valve for that engine and one of the control valves to each fan turbine is closed; the fans operate at half power. If a fan fails, the flow to that fan is shut off and the emergency jet adjacent to the failed fan is activated. The flow from the gas generators is split so that the thrust of the other fan and the jet balance.

As a first approximation to the guideline criteria, a thrust/weight ratio after failure of 1.1 was used. This permitted the installed thrust/weight ratio to be represented as a function of the number of engines, as shown in figure 16.

To compare the VTO and cruise thrust requirements, the airplanes were considered to have a constant L/D. In which case, the thrust required to cruise on two engines is independent of the total number of engines installed. Under static conditions, the total thrust/weight ratio installed to meet the cruise requirement is the thrust/weight ratio of the cruise systems multiplied by half the number of engines. The static thrust/weight ratio required for cruise is about 0.413.

A tabulated comparison of the VTO and cruise thrust requirements with schematics of the engine arrangements is shown in figure 17. It is apparent that six engines is a good design match. With larger numbers of engines the design is dominated by the cruise requirement, which has led to
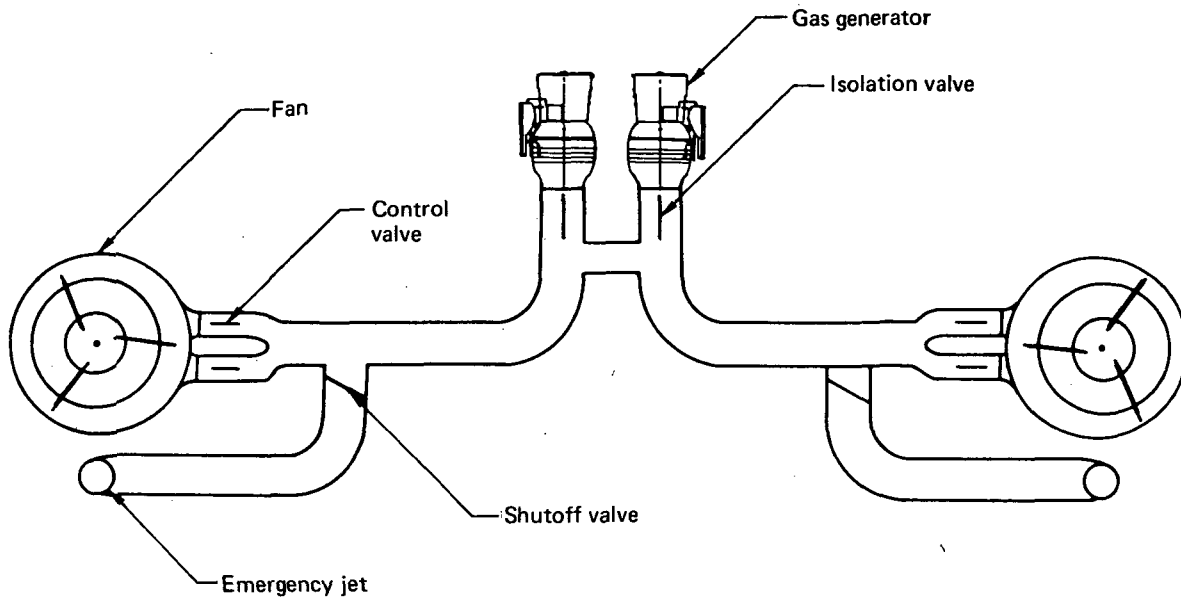
FIGURE 15.—INTERCONNECT SCHEMATIC
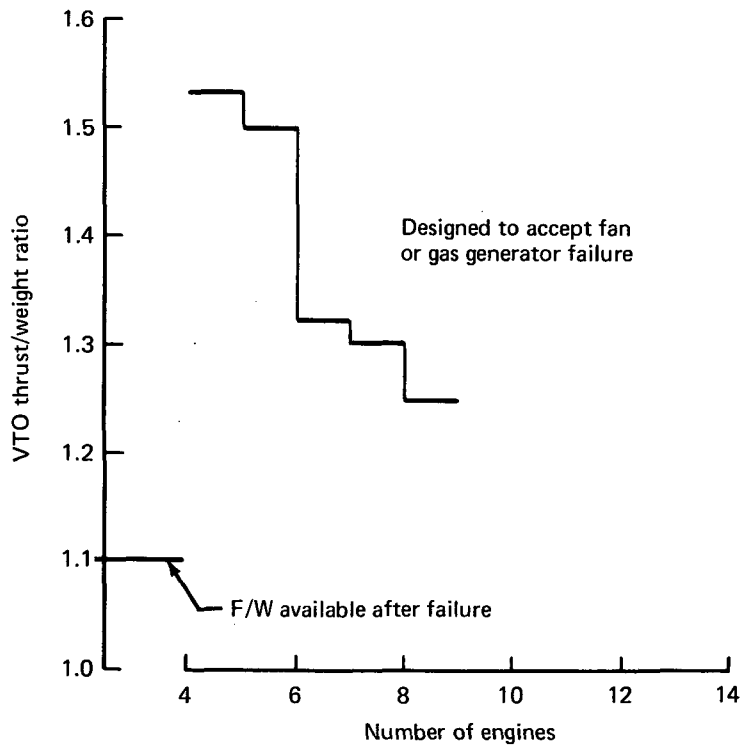
FIGURE 16.—TAKEOFF THRUST REQUIRED

marginal aircraft in the past. When the thrust installed is set by some requirement other than cruise, the resulting cruise system is generally good. The degree of mismatch should, of course, be small.
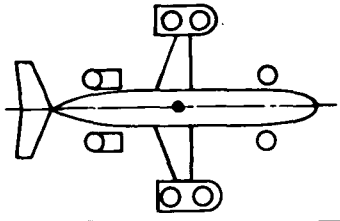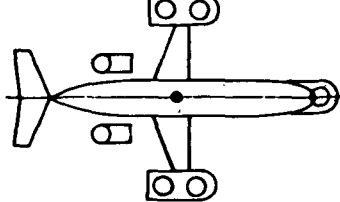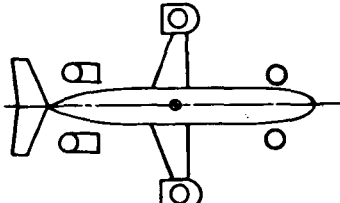
F/W required

		VTO	Cruise
Eight fans		1.24	1.65
Seven fans		1.30	1.44
Six fans		1.32	1.24

FIGURE 17.—CONFIGURATION SELECTION SUMMARY

This led to selection of a six-engine airplane for the baseline mission. A variety of six-engine arrangements were studied. The selected configuration is like that shown in figure 17. The forward lift fans are folded into the fuselage during cruise. Alternative six-engine airplanes were designed to avoid folding fans. In each case, a heavier and more cumbersome design resulted. A complete review of these configurations is contained in appendix C.

BASELINE CONFIGURATION

The six-fan, six-gas-generator configuration selected as the baseline was given the model number 984-139. The general arrangement is shown in figure 18, and an isometric rendition is shown in the summary (fig. 2).
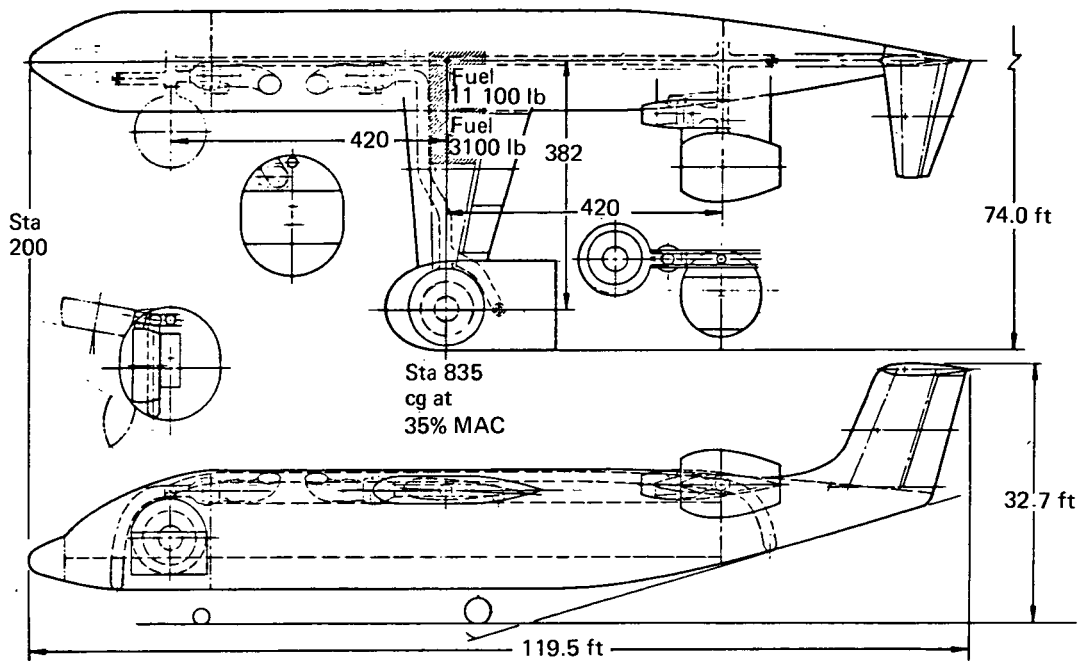
FIGURE 18.—MODEL 984-139 BASELINE

The airplane has a wing loading of 150 lb/ft^2, an aspect ratio of 5, and a thickness-to-chord ratio of 0.16. The empennage is designed to provide the required stability and control in normal flight. The vertical tail is swept 21.5°. Both horizontal and vertical tails have a thickness-to-chord ratio of 0.12.

The design is dominated by the propulsion system installation. The stowage of the front fans, the rotation of the cruise fans, and the arrangement of the ducts and valves in the interconnect system all contributed to the design.

PROPULSION SYSTEM INSTALLATION

The cruise fan is supported by two beveled roller bearings, which permits the fan to rotate between the cruise and the VTOL positions (fig. 19). Steel slip rings are used to seal the joint between the movable and stationary hot air ducts. The thrust and maneuver loads are transferred to the body by a carry-through box structure that straddles the duct.

The forward fan system is illustrated in figure 20. The fan rotates around a line on the upper fuselage. The cover doors move aft to deploy, minimizing the aerodynamic impact of opening and closing the doors.

The ducts and valves are designed to a maximum transient temperature of 2060° R and a pressure of 100 psi. A fail-safe duct is provided by use of double load-carrying walls. The duct size was set to limit the velocity to a maximum Mach number of 0.25. The most critical design condition is that of a fan failure. A schematic of the propulsion system is shown in figure 21, illustrating flow in the ducts for the condition of a fan failure.

In the example, the left front fan has failed; the flow to that fan is shut off. The diametrically opposite fan flow is limited to 40% of normal. All the other fans receive the full 100% and the nose emergency nozzle gets 160% of a gas generator output.

Configuration Match

The matched airplane has a maximum VTO gross weight of 126 300 lb and an operating empty weight of 94 750 lb. The propulsion system size is determined by the required VTO thrust/weight ratio of 1.37. The characteristic physical properties and performance properties are represented in figure 6 for the baseline VTO mission: 100 passengers, 400 nmi at M = 0.75.

Details of the weight and balance are contained in appendix D.

A parametric study was conducted to assess the impact of reducing wing loading from the assumed value of 150 psf. Figure 22 presents the results, at a thrust/weight ratio of 1.32, which show that wing loading may be reduced to 130 psf for a penalty in gross weight of only about 1%.

DESIGN MODIFICATION–CRUISE AT M = 0.85

The design changes required for cruise at M = 0.85 are illustrated in figure 23. The wing quarter chord is swept 30°; the wing-fuselage and fuselage-tail junctions are refaired through use of

TOP VIEW

SIDE VIEW

AFT VIEW

Static ground line
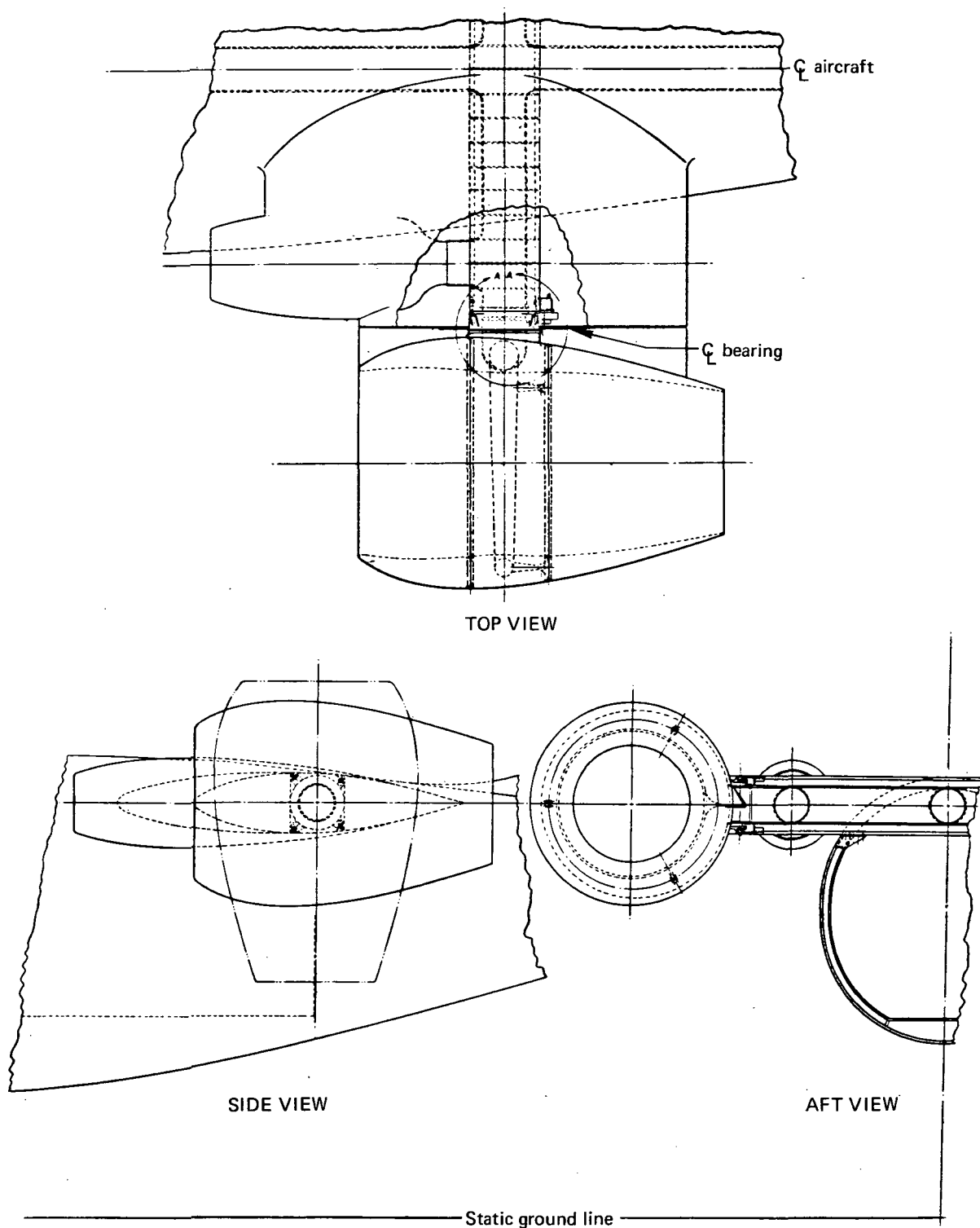
FIGURE 19.—CRUISE FAN MOUNT AND ROTATION—SIX-FAN AIRPLANES

26 796 lb thrust

Actuator

Static ground line
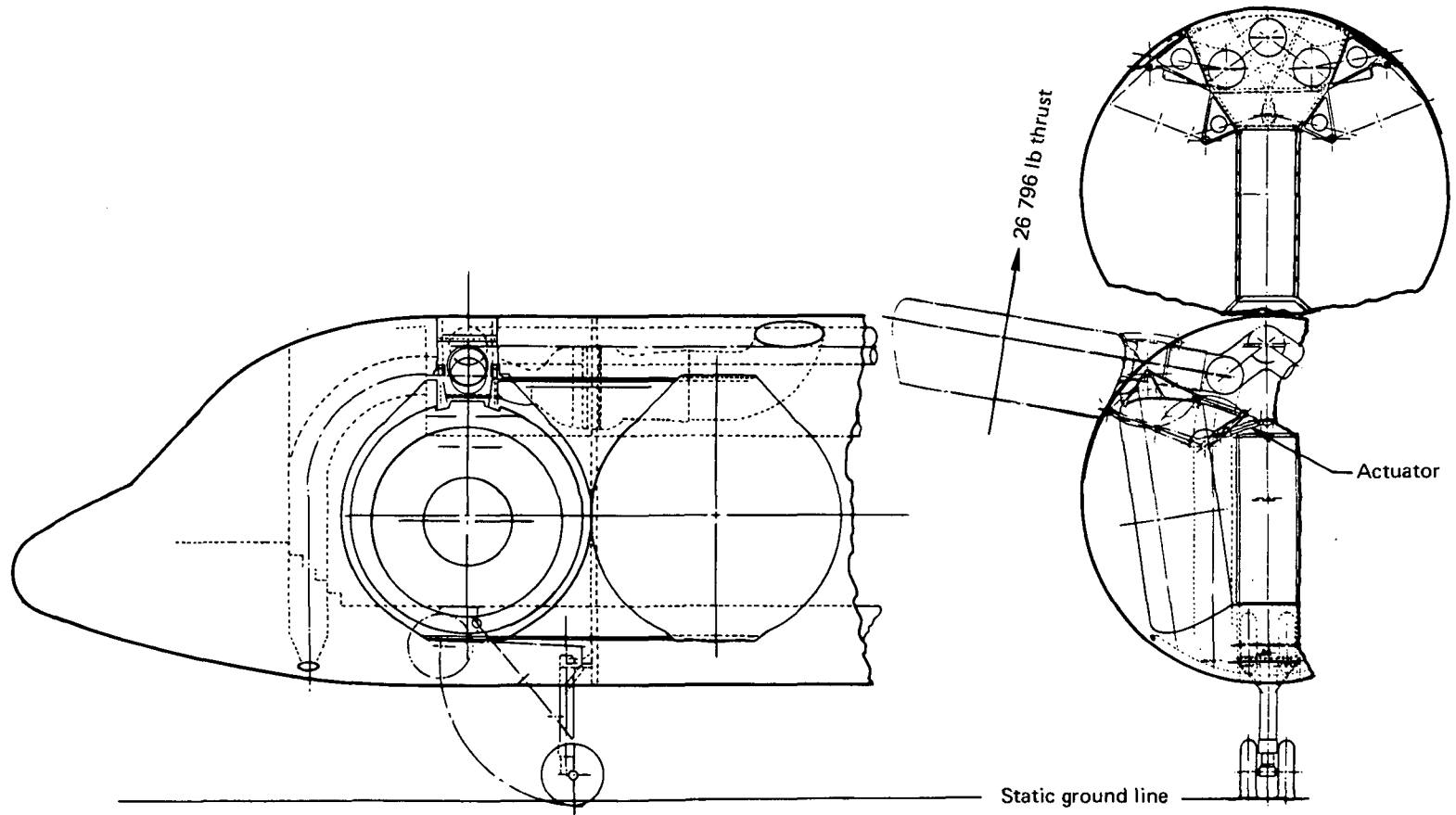
FIGURE 20.—FORWARD FAN SUPPORT AND ACTUATION—SIX-ENGINE AIRPLANES

100% flow = 85.9 lb/sec
at 2060° R and 10 787 lb/sq ft

160%

18.2 ID

17.6 ID

100%   100%

17.6 ID (all engines)

15.2 ID

100%   100%

17.6 ID

18.2 ID

100%   60%   40%

18.2 ID   17.6 ID

FIGURE 21.—MODEL 984-139 DUCT SCHEMATIC
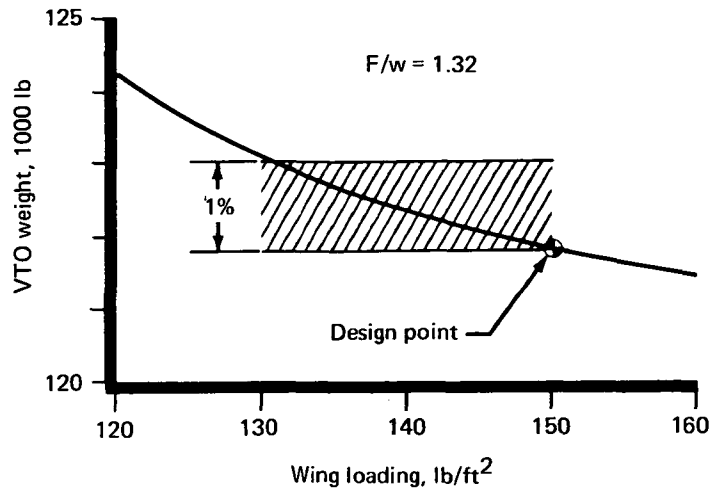
FIGURE 22.—MODEL 984-139 VTO WEIGHT VS WING LOADING
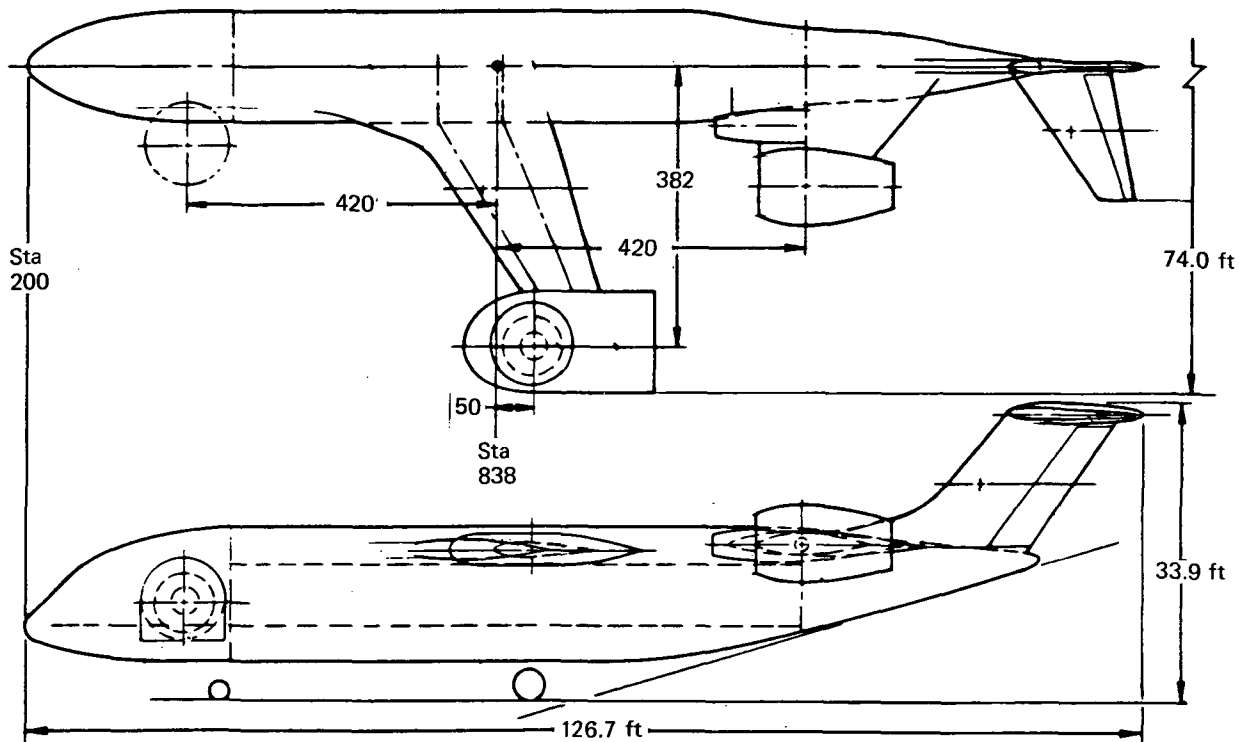
FIGURE 23.—MODEL 984-141 BASELINE MODIFIED FOR M = 0.85

advanced tailoring concepts; area rule principles are applied to the complete aft body to produce a fully integrated closure with minimum interference drag; and the cab shape is modified to eliminate rapid curvature changes and produce a smooth area distribution. A VTO weight of 132 800 lb matches the 400-nmi mission after these changes are made. This is an increase of 6500 lb over the M = 0.75 design.

A large number of wing sweep/thickness options are possible. Consideration of duct housing and wing weight favored the retention of the existing 16% normal thickness ratio. This led to the 30° sweep with a resulting streamwise thickness ratio of 14%. Adoption of advanced tailoring concepts in the region of the wing-body intersection and wing-lift fan pod juncture will probably involve the addition of a strake or "glove" in the wing root region in order to permit a reduction of sectional lift coefficient in this region. This will permit orderly "puddling" of the wing isobars while maintaining isobar sweep. Streamline contouring will be applied to the wing-mounted lift fan pods in order to maintain isobar sweep and sectional characteristics over the outboard wing.

The redesigned empennage will feature a horizontal tail with 30° quarter-chord sweep and 8.5% streamwise thickness ratio and a vertical tail with 40° sweep and a streamwise thickness ratio tapering from 12% at the root to 10% at the tip. An area ruled fairing will be added at the juncture of the two surfaces.

Incorporation of redesigned propulsive nacelles and struts will feature increased fineness ratio and inlet velocity ratio; the cowl shape will be designed using advanced transonic airfoil techniques. The nacelle struts will be increased in chord to maintain reasonable sectional thickness ratios.

DESIGN MODIFICATION—150 PASSENGERS

A 150-passenger version of the baseline airplane was designed and designated Model 984-145. A three-view drawing is shown in figure 24. The larger airplane is almost a direct scaling of the -139 (fig. 18). An eight-abreast seating arrangement was chosen as the best compromise. This resulted in a reduction in fuselage fineness ratio to 7.26 from the baseline value of 8.23, with the resulting inertias less than scaled. The thrust/weight ratio for VTO increased slightly due to the reduction in pitch moment arm. The size-matching characteristic lines are shown in figure 25. The 150-passenger and 100-passenger airplanes are compared. The shift in both performance and structural lines is due to the change in payload.

A comparison of pertinent features of the 100- and 150-passenger airplane is shown on table 1. The VTO gross weight increase from 126 300 lb to 192 500 lb, an increase of 52%, is indicative of the scaling.
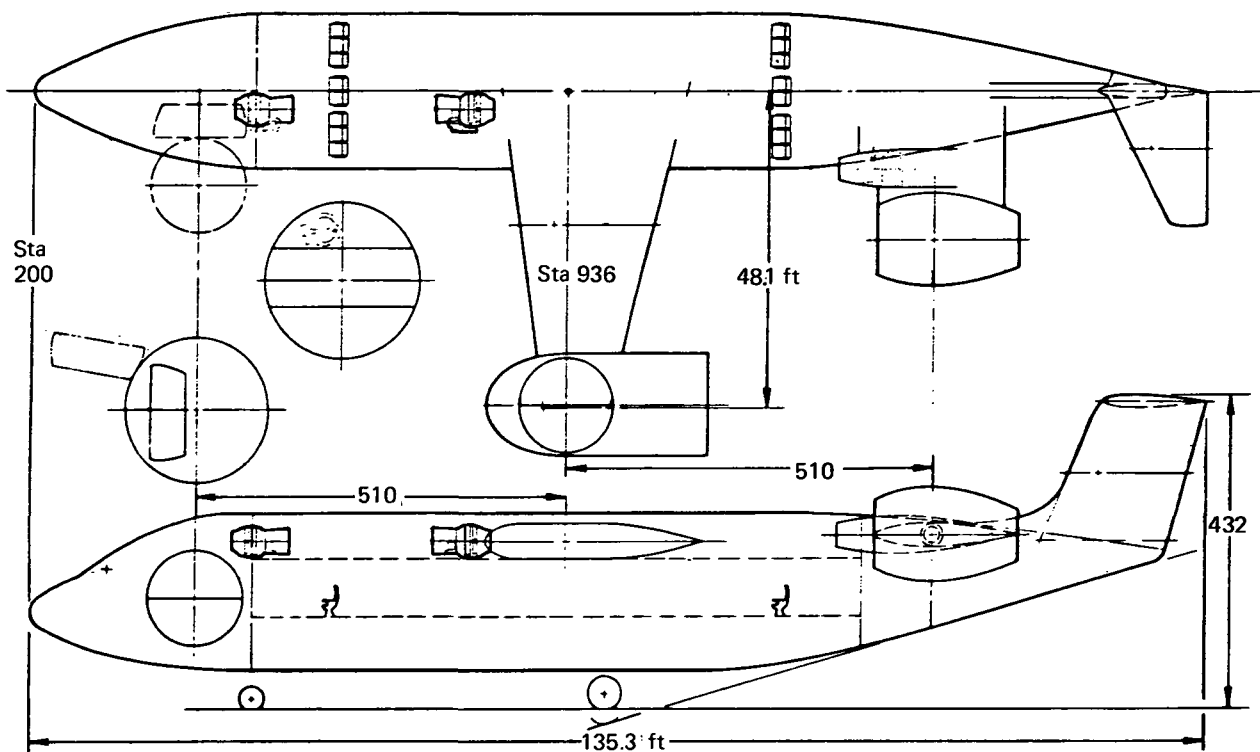
FIGURE 24.—MODEL 984-145 150-PASSENGER AIRPLANE
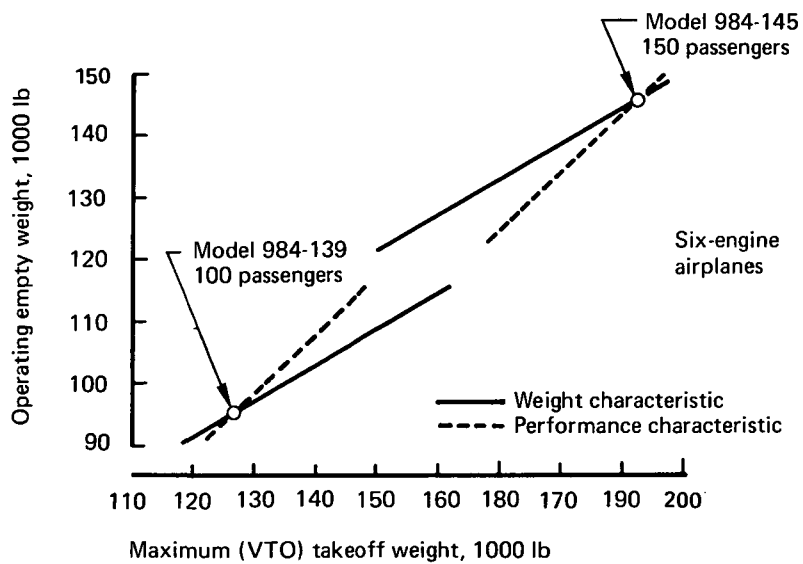
FIGURE 25.—SIX-ENGINE AIRPLANE WEIGHT COMPARISON—100 AND 150 PASSENGERS

TABLE 1.—COMPARISON OF 100- AND 150-PASSENGER AIRCRAFT

Items	Passengers	
	100	150
Floor area ratio	1.00	1.40
Body volume ratio	1.00	1.68
Fineness ratio	8.23	7.26
Pitch inertia ratio	1.00	1.90
Thrust/weight ratio	1.37	1.39
VTO gross weight	126 300 lb	192 500 lb

EFFECT ON DESIGN OF "NO FAN FAILURE"

The influence on the baseline airplane of a fan failure philosophy that assumes a fan will not fail during the flight was examined. The rationale behind this philosophy is similar to that used in wing design. It is assumed that a fan can be designed to have multiple loadpaths, means of wear and crack detection, and inspection techniques that would make the fans fail-safe as components. Under the assumption that fans can have operational reliability—will not fail in flight—the baseline airplane was modified. Since gas generator failure is still possible, the interconnect system is retained for power transfer and control. The emergency jet system is removed and the VTO thrust/weight requirement is set by gas generator failure and is reduced to 1.30 from 1.37.

Baseline Mission—"No Fan Failure"

The reduction in installed thrust and the removal of the emergency jet system results in a weight reduction. The shift in the weight characteristic line due to these changes at the baseline gross weight of 126 300 lb consists of:

$$
\begin{aligned}
&\text{Baseline OEW} && = 94\ 750\ \text{lb} \\
&\quad \Delta\text{OEW emergency jet system} && = \text{-}1350\ \text{lb} \\
&\quad \Delta\text{OEW reduce F/W to} \\
&\quad 1.30\ \text{from } 1.37 && = \text{-}2180\ \text{lb} \\
&\quad \Delta\text{OEW total} && = \text{-}3530\ \text{lb} \\
&\text{New OEW} && = 91\ 220\ \text{lb}
\end{aligned}
$$

The new characteristic line, parallel to the original, is shown in figure 26. The performance characteristic line is changed only slightly. The airplane with no fan failure matches at a gross weight of 115 800 lb, which is a weight reduction of 8.3%.

Weight characteristic
Performance characteristic

Six-engine
baseline
model 984-139
(design for fan
failure)

400 nmi

200 nmi

Model 984-139 (no fan failure)

Model 984-142 (no fan failure)

Operating empty weight, 1000 lb

Maximum (VTO) takeoff weight, 1000 lb
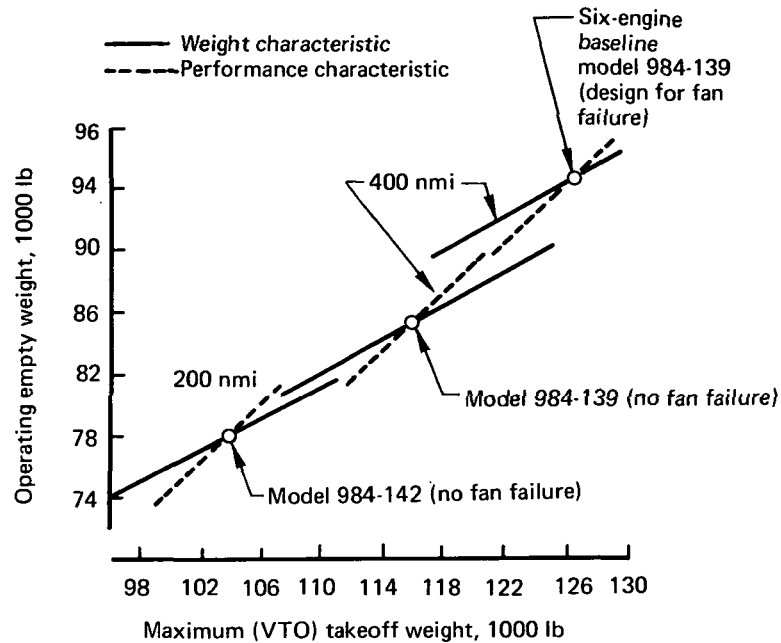
FIGURE 26.—SIX-ENGINE AIRPLANE WEIGHT COMPARISON

Short-Range Mission—200 nmi

The effect of range reduction was examined for the airplane with the "operationally reliable" fans. A design for a 200-nmi VTOL airplane without STOL overload capacity was made as a modification to the baseline airplane with "no fan failure," Model 984-142. In addition to the range reduction, the design benefited slightly from a reduction in sink speed, baggage allowance, and galley requirements. A double aisle was added to speed passenger handling. This is a weight increase.

The baseline weight characteristic line was shifted again by these changes. For the baseline gross weight of 126 300 lb, the incremental changes in operating empty weight are:

Baseline OEW		= 94 750 lb
Fail-safe OEW from baseline mission		= 91 220 lb
Δ OEW—STOL overload	= -1250 lb	
Δ OEW—sink speed, cargo	= -360 lb	
Δ OEW—double aisle	= +400 lb	
Δ OEW total	= -1210 lb	
OEW for 200-nmi design (model 984-142)	= 90 010 lb	

This new characteristic line is also shown in figure 26. The change in range caused an appreciable shift in the performance line. The six-fan airplane designed for "no fan failure" and 200-nmi VTOL range has a VTO gross weight of 103 700 lb.

Cruise Speed Reduced to M = 0.65

The effect on the 200-nmi "no fan failure" design of reducing cruise speed from M = 0.75 to M = 0.65 was studied with a view to better economy and simplicity of design.

The effect on wing weight of increasing the thickness from 0.16 chord to 0.20 chord was examined. The results are shown in figure 27. Within the range of interest, there is less than 1% change in wing weight.
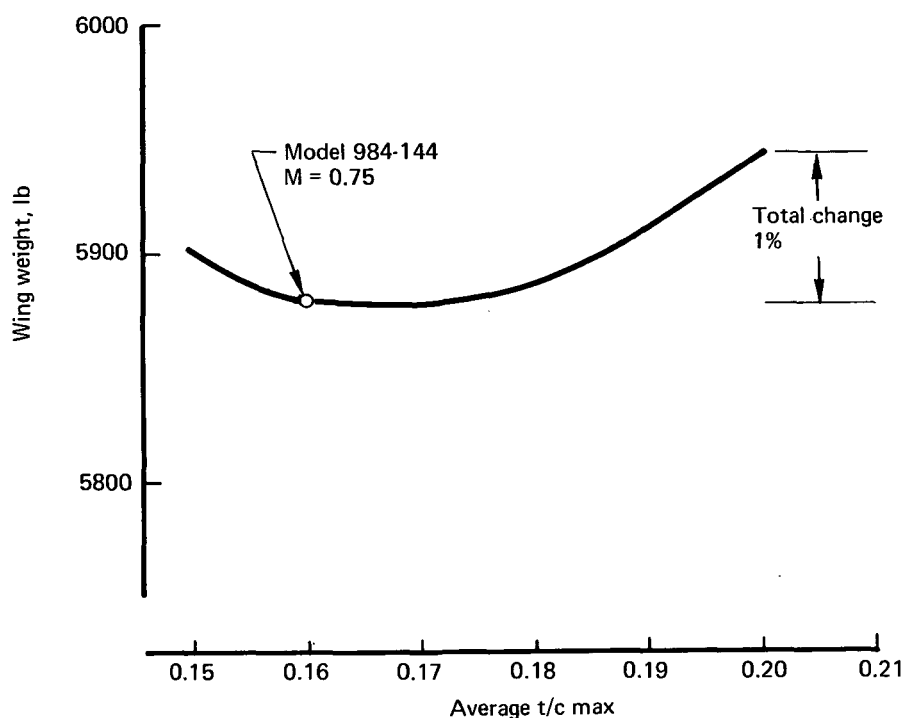
FIGURE 27.—WING THICKNESS TRADE FOR MACH = 0.65 STUDY

To increase the M = 0.65 cruise L/D, a "nonpeaky" airfoil section was chosen. There is a slight increase in L/D at speeds below the drag rise Mach number although there is also a reduction in drag rise Mach number. The overall change in L/D may be seen from a comparison with the baseline airplane L/D curve in figure 28. Increasing the thickness ratio to 20% reduces the Mach number for maximum L/D to 0.68.
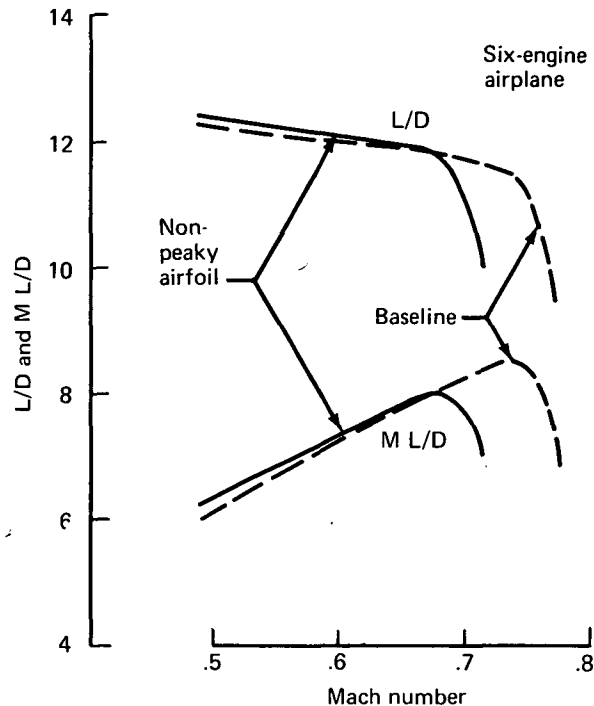
FIGURE 28.—MODIFICATION TO M = 0.65

As a result of these changes, a reduction in mission fuel, not counting reserves, of about 1% is possible. This small saving could be realized by operating the M = 0.75 design at M = 0.65, without any redesign.

FOUR-FAN, FOUR-GAS-GENERATOR AIRPLANE

A four-fan, four-gas-generator airplane is feasible with the concept of no fan failure. The thrust/weight ratio required under this rule is compared with the requirement to tolerate a fan failure, figure 29. With operationally reliable fans, the thrust/weight ratio required with four units is the same as that required with six when a fan failure must be tolerated. These data, like those of figure 16 from which the design for fan failure curve is taken, are based on achieving a thrust/weight ratio of 1.1 after failure. They do not account for the moment arm differences and other detail design features that modify these values on actual configurations.

Configuration Description

The best four-engine design is shown in figure 30. The two cruise fans are mounted under the wing. They rotate around the pylon for vertical flight. Isometric views are shown in the summary,
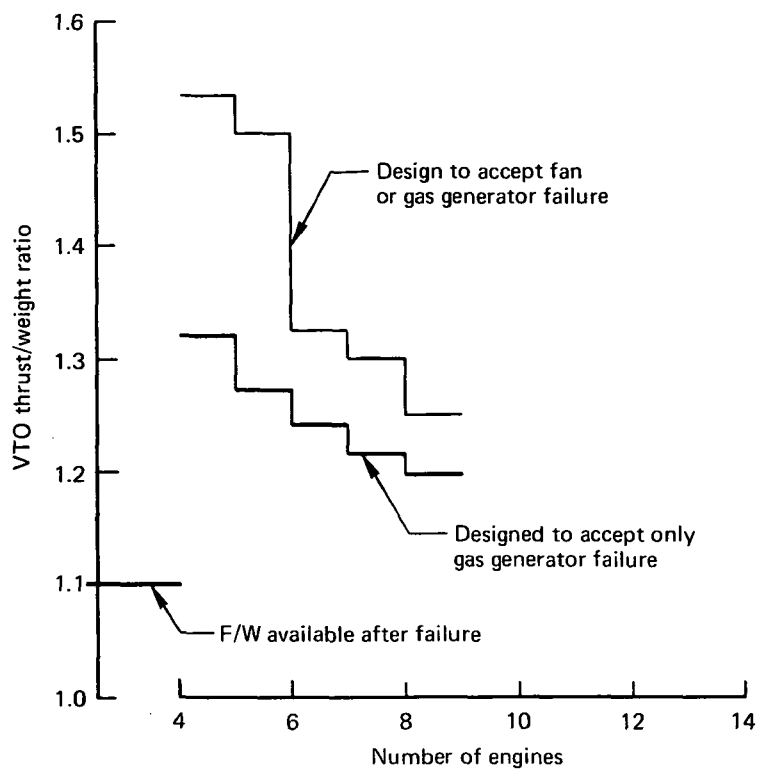
FIGURE 29.–THRUST/WEIGHT REQUIRED
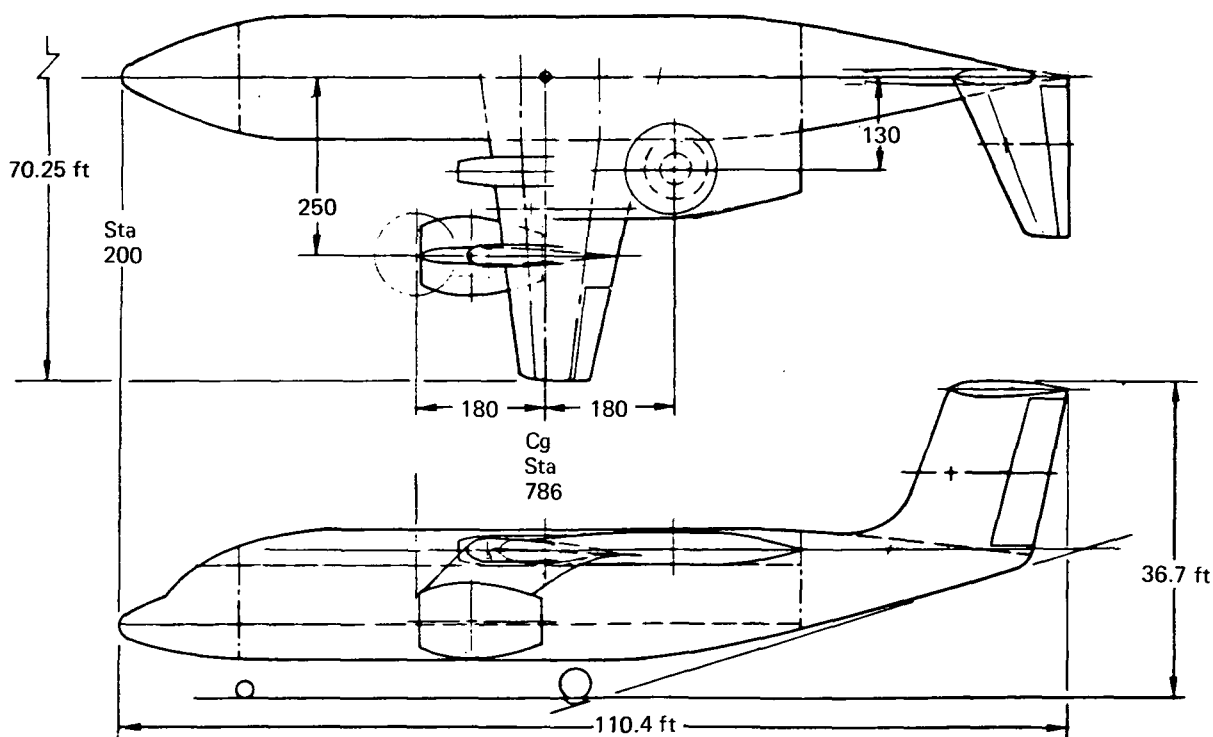
FIGURE 30.–MODEL 984-144 FOUR-ENGINE AIRPLANE

figure 2. The lift fans are contained in trailing edge extensions. The lift fan thrust is rotated with louvers. All four fans are connected, and the interconnect system is comparatively short, having only a small amount of ducting over the fuselage.

Two possible methods for rotating the cruise fan were designed. The first uses a flexible section of duct; this scheme is shown in figure 31. The other method employs rigid ducting with oblique rotation bearings. This latter scheme is shown in figure 32. Both schemes are reasonable.

Four-engine airplane
Model 984-144

Bellows

Rigid duct

Hydraulic motor
(45 hp required)

Static ground line
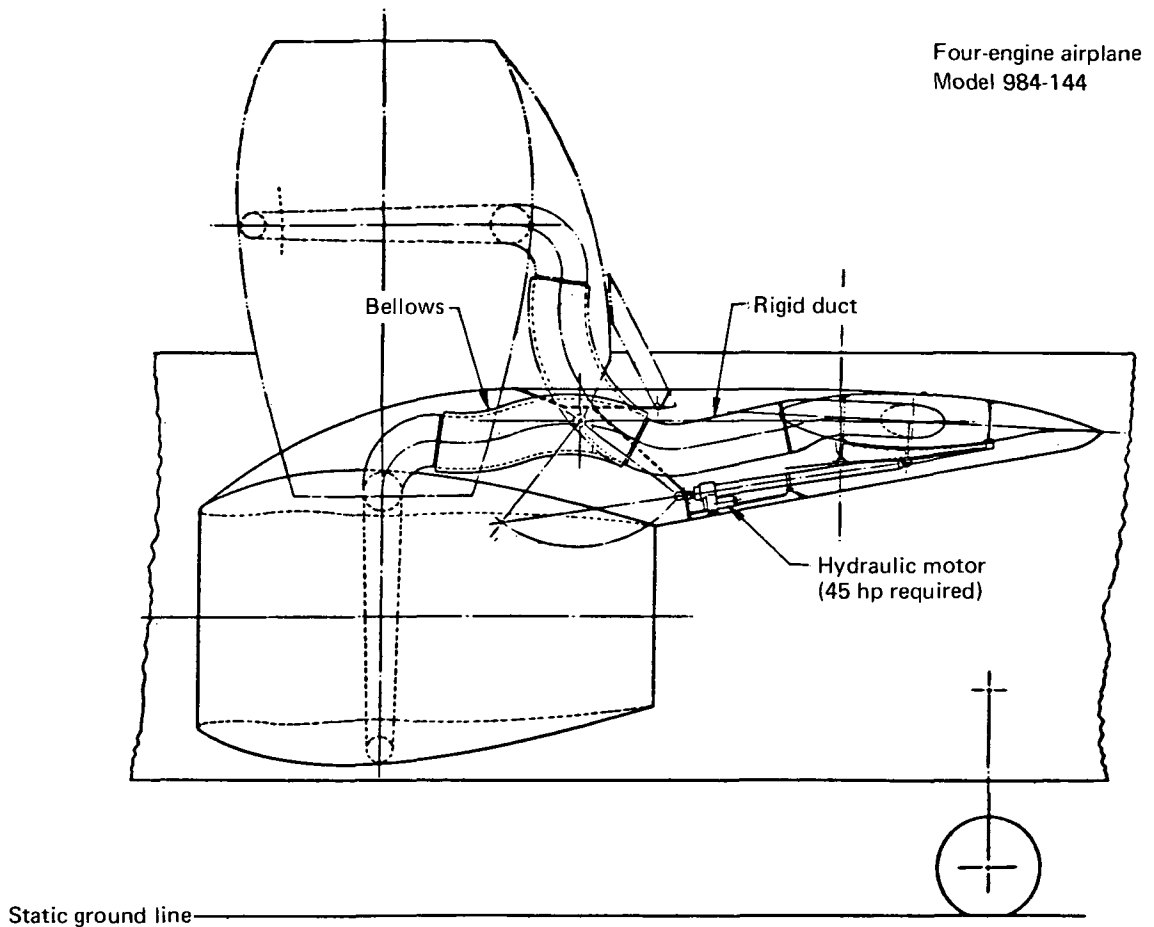
FIGURE 31.—CRUISE FAN ACTUATION—FLEXIBLE DUCT

38

Four-engine airplane
Model 984-144

Hydraulic motor
(45 hp required)

Static ground line
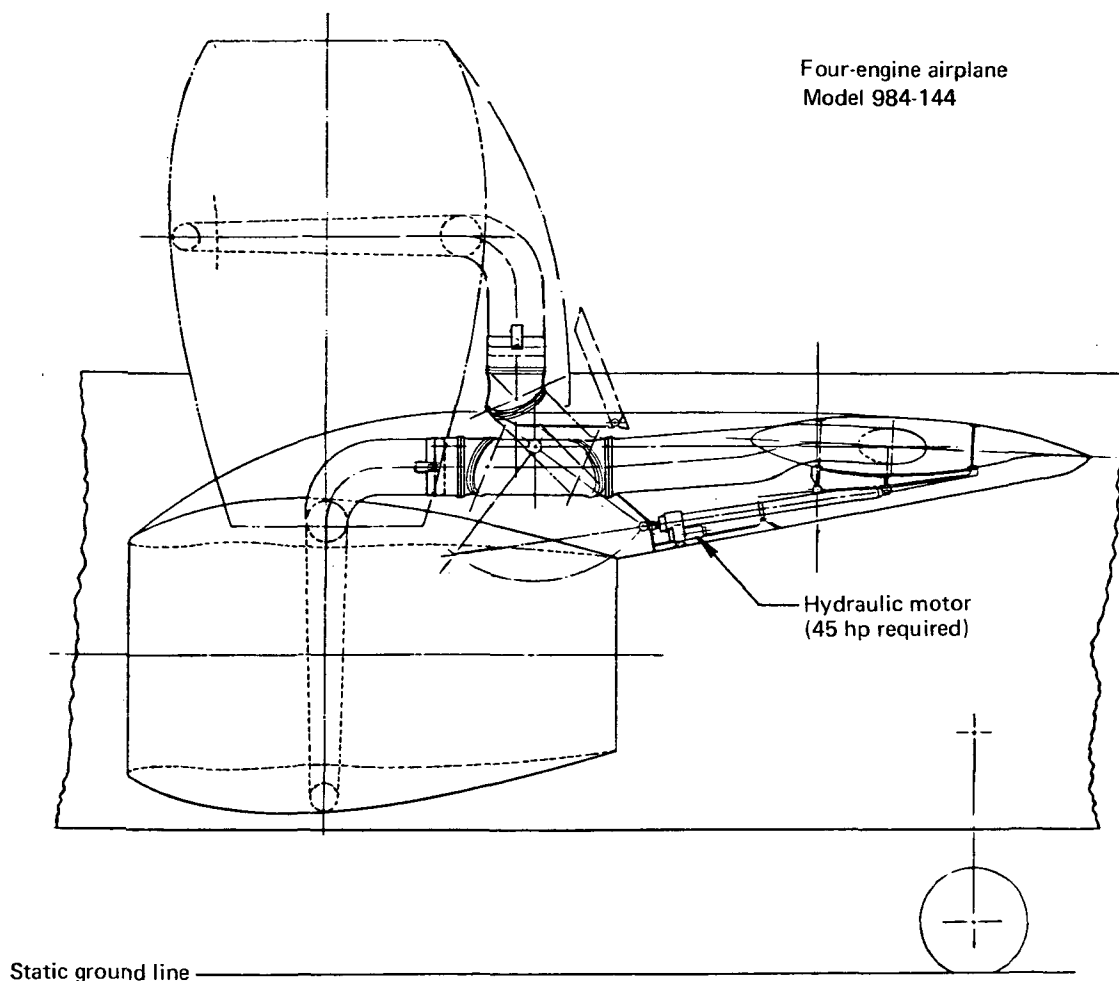
FIGURE 32.—CRUISE FAN ROTATION OBLIQUE CUTS

The wing loading is 125 lb/ft^2. This reduction from 150 lb/ft^2 resulted from the engine arrangement. The engine, fan, and duct space requirements led to the increased area. However, the airplane can fly at 250 kn at 10 000 ft, without buffet and without flaps, at this reduced wing loading. Adding flaps to this wing would considerably complicate the design.

The resulting cruise L/D is shown in figure 33. In comparison with the six-engine design, there is a decrease in aerodynamic efficiency due mainly to the increased wetted area.

The thrust loading for this airplane is determined differently for the fans and gas generators. For the fans, which do not fail, the maximum loading occurs with all engines operating when the
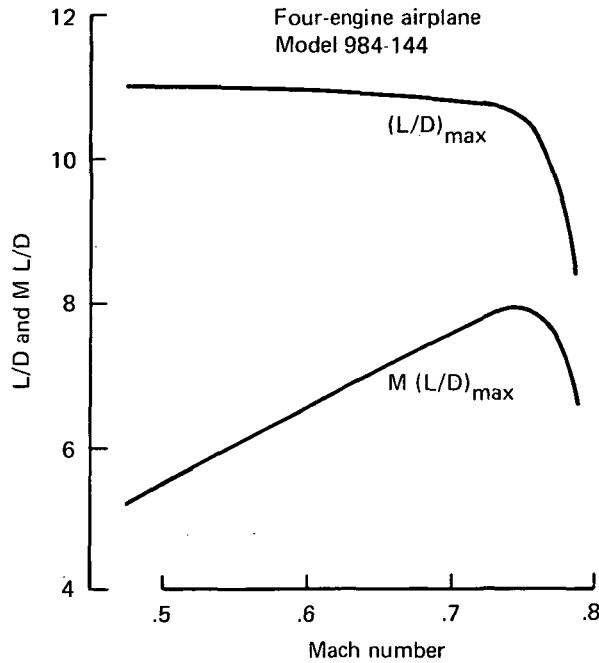
FIGURE 33.—AERODYNAMIC EFFICIENCY

total thrust must equal 1.1 times the weight plus trim and applied control. After failure of a gas generator, the total thrust requirement reduces to 1.05 times the weight plus trim and control.

Putting this symbolically with all engines operating

$$F_{tot} = 1.1W + \Delta F_{trim} + \Delta F_{applied\ control} \tag{4}$$

After a failure, the equation becomes

$$F_{tot} = 1.05W + \Delta F_{trim} + \Delta F_{applied\ control} \tag{5}$$

This is the same as (1), shown in "Design Guidelines" section. For the fans, the design condition is with all engines operating, equation (4). The actual thrust/weight ratio needed to meet that requirement is 1.27.

The gas generator is sized by the failure requirement. Three gas generators driving four fans must produce the required thrust. Four of these gas generators could, if the fans were large enough, produce a thrust/weight ratio of 1.49. Since the fans were designed to an F/W of 1.27 the gas generators will operate at part power except in the event of a gas generator failure.

Airplane and Performance Match

The basic weight differences between the six- and four-engine design result from the thrust/weight ratio differences, removal of the forward fuselage cutout, and the reduction in wing loading. At the VTO gross weight of the six-engine, 200-nmi airplane of 103 700 lb, the following increments in operating empty weight caused the shift in the weight characteristic line shown in figure 34.

$$
\begin{array}{llr}
\text{Six-engine OEW} & & = 78\ 000\ \text{lb} \\
\Delta\ \text{OEW wing area} & = & +50\ \text{lb} \\
\Delta\ \text{OEW fwd body cutout} & = & -1210\ \text{lb} \\
\Delta\ \text{OEW engine size increase} & = & +2340\ \text{lb} \\
\Delta\ \text{OEW total} & = & +1180\ \text{lb} \\
\text{Four-engine OEW} & & = 79\ 180\ \text{lb}
\end{array}
$$
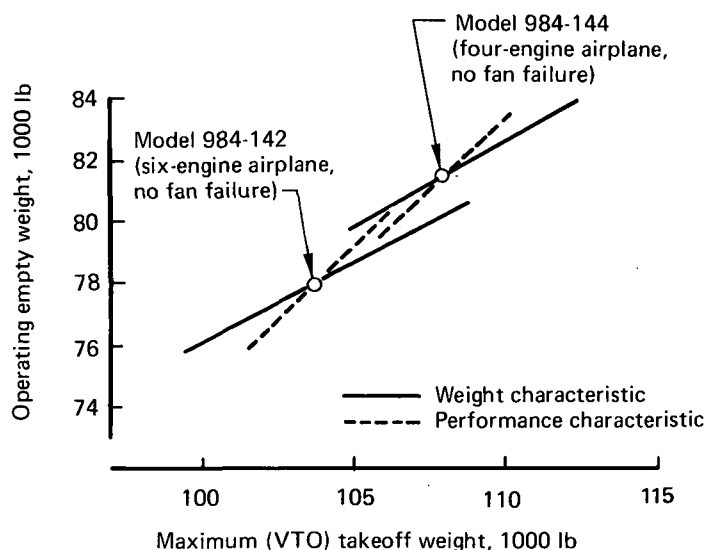
FIGURE 34.—FOUR- AND SIX-ENGINE AIRPLANE WEIGHT COMPARISON

The shift in the cruise characteristic line in figure 34 resulted from the increased cruise thrust and lower L/D of the four-engine airplane. The installed thrust/weight ratio of 1.27 divided among four engines, of which two are used in cruise, results in a thrust/weight ratio available for cruise of about 0.63 compared to the approximate 0.413 requirement of the baseline selection. The resulting off-design operation is a factor in the shift of the performance characteristic. The gross weight of the four-engine airplane is 107 900 lb, a 4% increase over the six-engine design.

The four-engine design has exceedingly good climb characteristics that, together with the lower number of engines and the removal of the fuselage cutout, cause this airplane to be more economical than the slightly lighter six-engine design. In addition, the operators' subjective feeling for four engines as against six, makes this concept attractive.

STABILITY AND CONTROL

V/STOL lift fan aircraft operate in a flight regime which extends from hover to maximum cruise speed. Stability and maneuverability must be satisfactory for commercial operation over the entire regime. The means of control differ at the different ends of this range. In hover, the aircraft is supported and maneuvered with thrust from its fans. During conversion to normal flight, both powered and conventional aerodynamic controls are used and transition is complete at the speed at which aerodynamic lift and control is sufficient.

CRUISE STABILITY AND TAIL SIZING

The stability in conventional flight was treated lightly. The tail arms and areas were determined in order to properly define the airplane for the weight and cost analysis. Horizontal and vertical tail sizes were chosen on the basis of comparison with existing V/STOL aircraft. For the V/STOL aircraft the tail volume correlates well with gross weight whereas the tail volume coefficient does not. This is due to the fact that the volume coefficient has been nondimensionalized by the wing area and tail length. The wing area of a V/STOL airplane is designed for cruise rather than takeoff and landing and therefore may bear little relation to stability or controllability. Data on horizontal and vertical tails are shown in figures 35 and 36. The necessary control, provided by the area and length of the tail surfaces, will reflect the airplane's inertias; the linear trend shown appears to follow the "square-cube" law of vehicle sizing.

For comparison, several points for pure STOL aircraft are shown in these figures, and again show a linear correlation, though with a different slope.

Treating the tails in this manner handles, at a stroke, all the variables and considerations that would be studied in a more detailed design process. These include the contribution of the aerodynamic surfaces to transition and conversion, STOL landing safety, and dynamic stability.

The flap system used for the low speed approach flightpath study is a double-slotted flap with a deflection of 7°/27° during conversion and approach. Flap span is that remaining after sizing the ailerons. The aileron span of 0.19 b is necessary to meet the critical roll requirement at 10 000 ft and 200 kn indicated air speed.

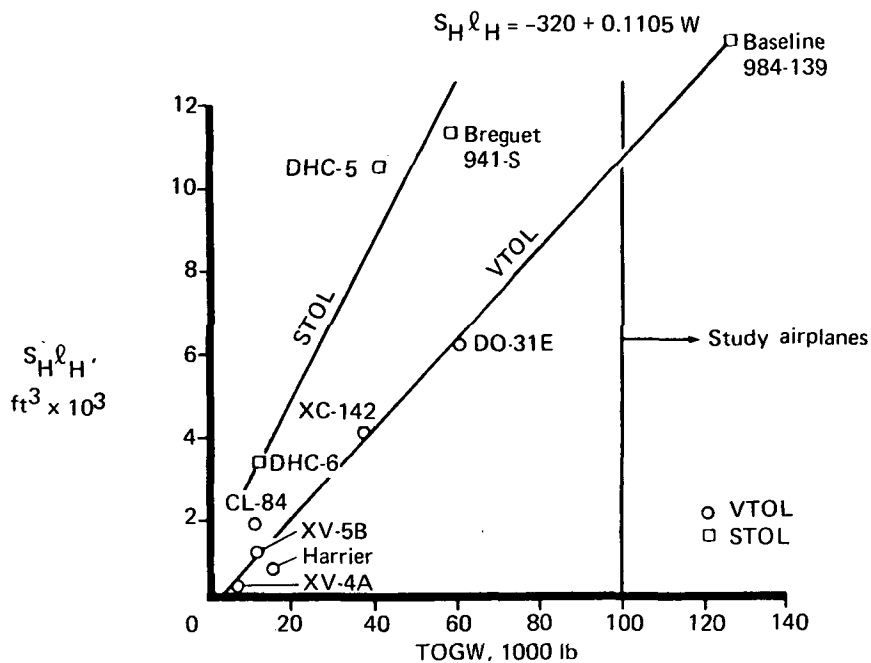$$S_H \ell_H = -320 + 0.1105\,W$$

FIGURE 35.—VARIATION OF HORIZONTAL TAIL VOLUME
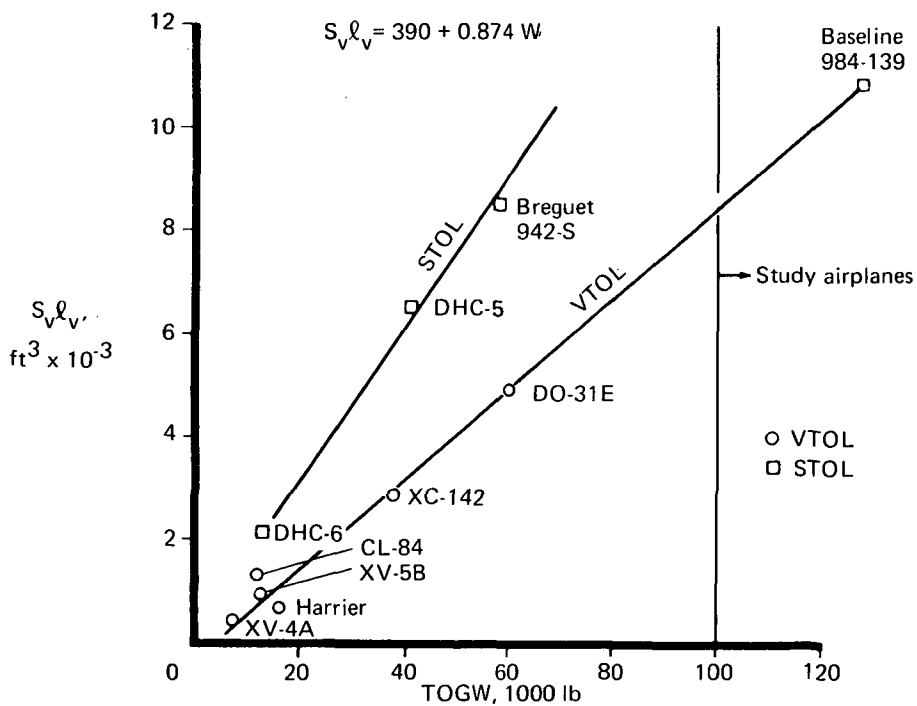WITH TOGW FOR VARIOUS AIRCRAFT

$$S_v \ell_v = 390 + 0.874\,W$$

FIGURE 36.—VARIATION OF VERTICAL TAIL VOLUME WITH TOGW
FOR VARIOUS AIRCRAFT

PERFORMANCE ON STEEP GLIDE SLOPES

The performance characteristics of the six-fan baseline aircraft along steep glide slopes was determined. The range of flightpath angles considered extends from level flight ($\gamma = 0°$) to a 30° glide slope ($\gamma = -30°$). The thrust levels and thrust vector angles associated with various steady-state flight conditions were also found. The results are presented in terms of speed and flightpath angle ($V - \gamma$) plots, similar to $V - \gamma$ plots of conventional aircraft except that power level is shown as gross thrust/weight ratio instead of throttle setting, and pitch attitude is held constant; the thrust vector angle varies as a parameter along the constant thrust/weight ratio lines.

The acceleration parallel and normal to the flightpath that could be achieved by power application, a point of interest for maneuvering and decelerating for landing, was determined.

Time histories of thrust and thrust vector angle manipulations during various descent maneuvers were studied to identify possible operational limitations.

Steady-State Conditions

The relationship between speed and flightpath angle, under equilibrium conditions, was determined as functions of thrust/weight ratio and thrust direction. The thrust vector angle (σ) is measured from the airplane centerline. The convention is illustrated in figure 37. When the thrust is perpendicular to the centerline, $\sigma = 90°$; when it is along the flight axis, as for cruise, $\sigma = 0°$. Only the two aft lift/cruise fans can be rotated to $\sigma = 0°$. Practical limitations to thrust vectoring on the lift engines led to the assumption of a capability of $\Delta\sigma = 30°$ from the vertical; $60° \leqslant \sigma \leqslant 120°$. During approach this limit was applied to all the fans.

Equilibrium values of γ and V for various thrust levels are shown on figures 38 and 39. They are for level attitude and 10° nose-down attitude conditions. Superimposed on the plots are lines of constant thrust vector angle. At a constant thrust/weight ratio, the glidepath angle and velocity vary with thrust vector angle. For example, in figure 38, which is for a level attitude ($\theta = 0°$) with F/W = 0.7 and $\sigma = 75°$, steady-state flight is at 120 kn and $\gamma = -3.6°$; then at $\sigma = 80°$, the condition for equilibrium is 90 kn at $\gamma = -8.8°$. With the airplane level, the angle of attack increases as the glide slope steepens; $\alpha = -\gamma$; with stall occurring at about 19°. In the stall region, two glide slopes are possible for a given thrust level and angle. At F/W = 0.7 and $\sigma = 83°$, we can have V = 77 kn and $\gamma = -14°$ or V = 68 kn and $\gamma = -24°$. The latter condition is with a stalled wing and should be avoided.

Flight past stall at appreciable speed is not practical for reasons of buffet and danger of spin. This region in figure 38 corresponds to rates of sink between 2000 and 5000 ft/min, which certainly are not appropriate for final descent maneuvers. A useful or safe γ -V region is shown on the figure.
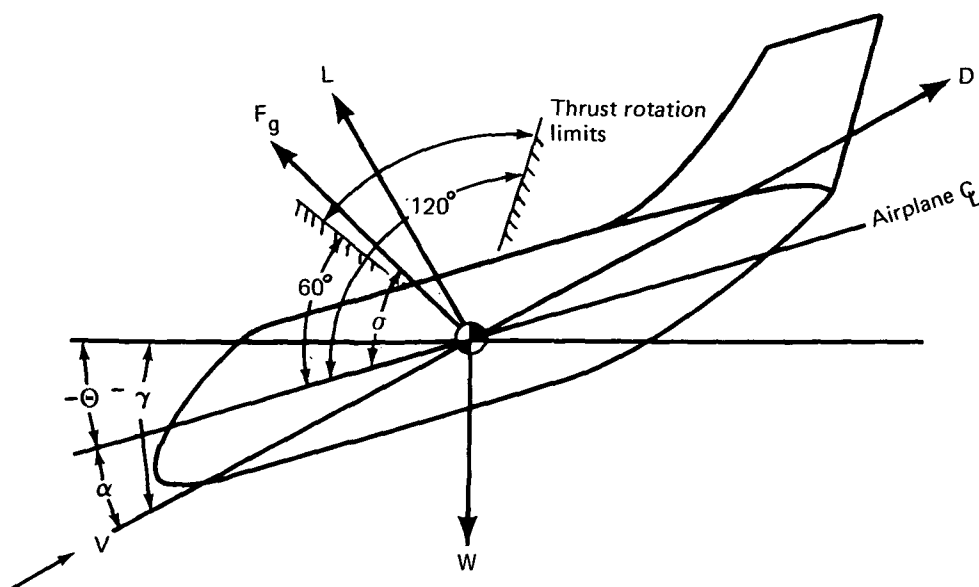
FIGURE 37.—POWERED LIFT AIRCRAFT ON FLIGHTPATH

The area shown is an approximation to the region of buffet-free flight without danger of spin. The dashed portion of the operating envelope indicates an area where flight at low altitude should be avoided.

It is also seen from this figure that small changes in thrust level or vector angle at low speed and near stall can cause large changes in steady-state glide slope angle; this is another reason for avoiding the stall region and suggests that means for precise control of V and γ must be provided.

Figure 39 presents the same information with the aircraft in its maximum allowable nose-down pitch attitude of $\theta = -10°$. The region of most favorable flight conditions has moved to higher speeds and steeper glide slopes, and the troublesome stall region has moved off the plot. The aircraft is now capable of very high descent rates, which might be useful for the initial portions of the landing approach, without encountering the stall region. Note, however, that level flight requires much higher power levels than in the previous case. The nose-down attitude is more favorable for flightpaths where $-10° > \gamma > -20°$.

Minimum γ points of the constant σ contours correspond to points of maximum vehicle L/D; they are characterized by the fact that both increases and decreases in power result in steady-state flight at shallower glide slope angles with σ and θ held constant.
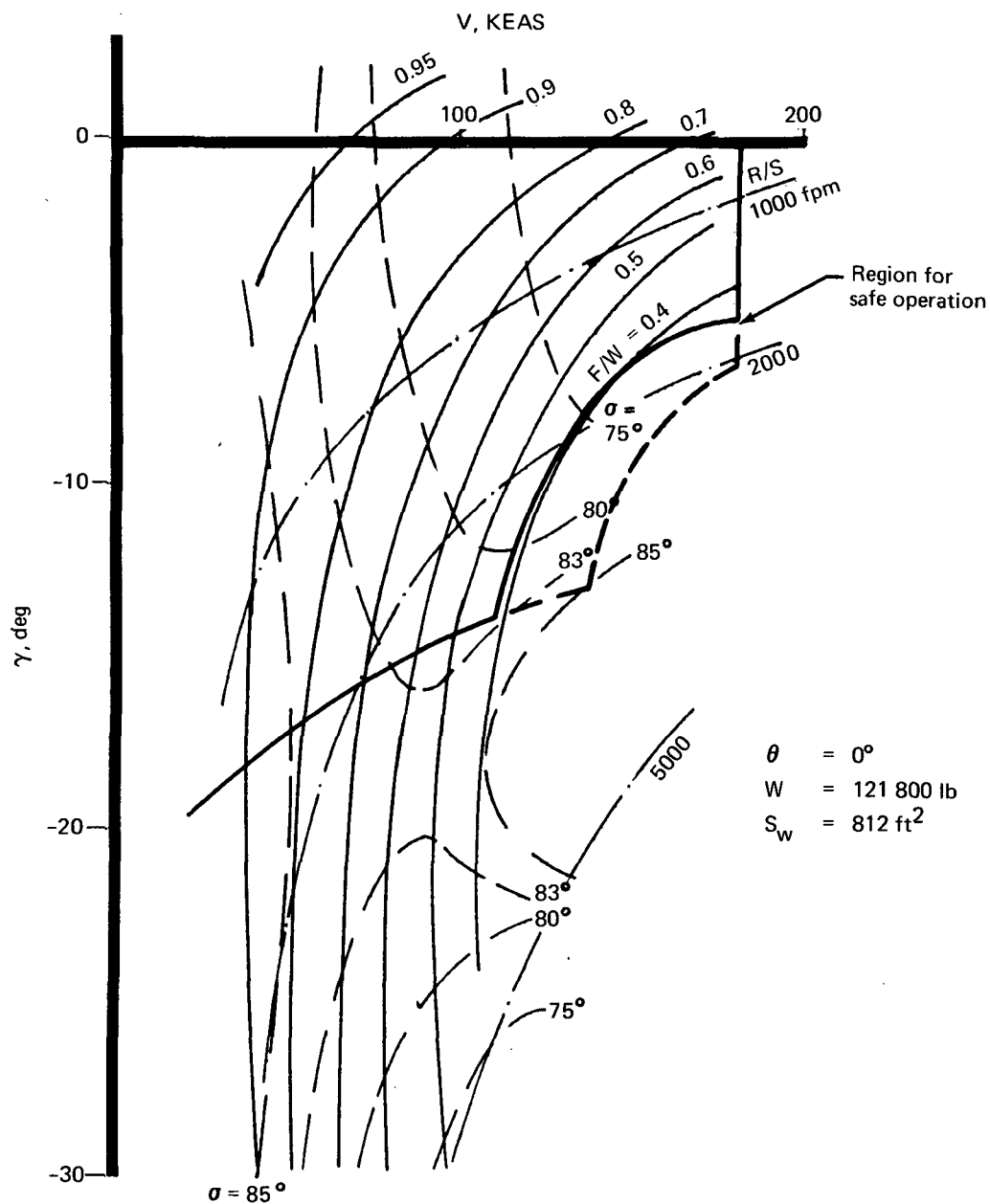
FIGURE 38.—DESCENT PERFORMANCE, Θ = 0

47
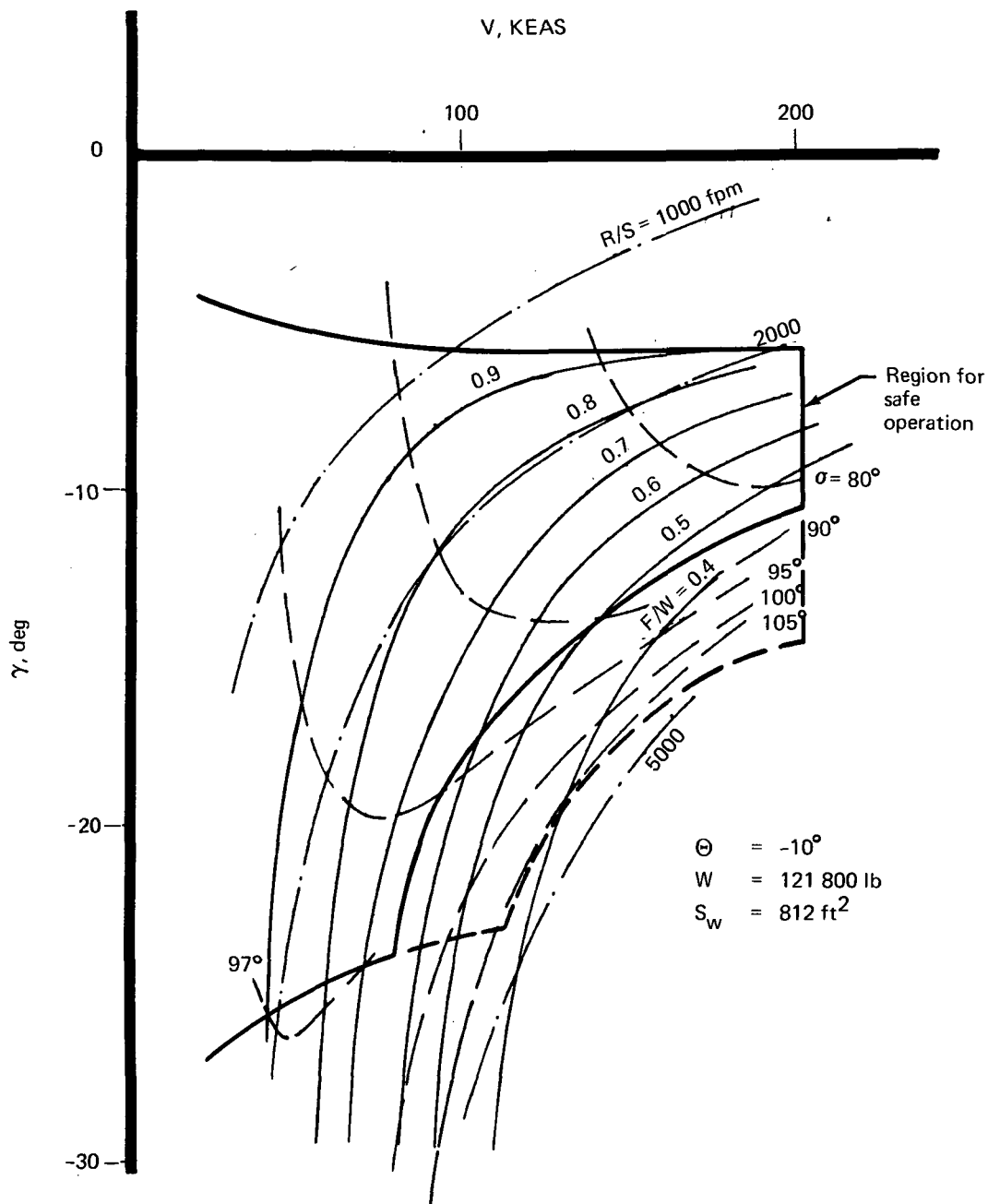
FIGURE 39.—DESCENT PERFORMANCE, $\Theta = -10°$

The shape of the constant thrust angle curves at speeds below that for minimum   have the appearance associated with flight along the "backside" of the drag polar. The flightpath is relatively insensitive to thrust level, but would respond to changes in thrust angle.

From figures 38 and 39, the range of approach paths available, without the restriction of a fixed attitude, is seen to be large. A letdown can be started at a high descent rate with the nose down and completed gently with the fuselage level.

Propulsive Maneuvering Capability

The ability to accelerate and decelerate with thrust alone is important at the low speeds associated with the end of the approach. At higher speeds, the aerodynamic forces are large and maneuvering capability does not rely on thrust alone. Examples of the accelerations available at a constant attitude along lines of constant flightpath are shown by figures 40 and 41. The accelerations normal to the flightpath are in figure 40 and those along the flightpath in 41. Both positive and negative normal accelerations are available. As the glide slope becomes steeper, the upward acceleration capability increases and the downward capability decreases, up to the flightpath angle at which the stall region is entered. The angle of attack and wing lift increase; this frees more vehicle thrust for upward maneuvering. At the same time, wing lift cannot be removed as can thrust so the downward maneuvering capability decreases. The variation of the curves with speed reflects the varying amount of weight carried on the wings; as V decreases, more load is carried by thrust, and the thrust available for upward maneuvering decreases. The rapid decrease of a_n (+) with V for $\gamma = -30°$ occurs because the equilibrium vector angle approaches the forward vector limit; hence, there is little margin for maneuvering without upsetting the drag balance. At $V = 0$, the upward acceleration equals the design F/W of the aircraft, and the downward acceleration depends on how much thrust can be spoiled; here this was taken to be 80%.

Figure 41 shows the ability to accelerate or decelerate along the flightpath; this ability is strongly dependent on the amount of lift being carried on the wings and on the physical limitations of thrust vectoring. Forward acceleration is seen to be always limited by the forward (60°) vector limit; it decreases with increased speed or steeper flightpath angles. Rearward acceleration, a_p (-), is larger generally because this vector limit (120°) is opposite to the thrust direction needed for steady flight and ample thrust is available for lift balance. Thrust limitations caused by the lift balance rapidly worsen at high speeds, however, with the result that the ability to decelerate while maintaining glide slope and attitude quickly approaches zero as speed is increased. Thus, it may not be desirable to establish a high-speed glide slope before decelerating to land. The aircraft should first slow to a speed at which it has adequate deceleration capability along the glide slope and then begin its descent.
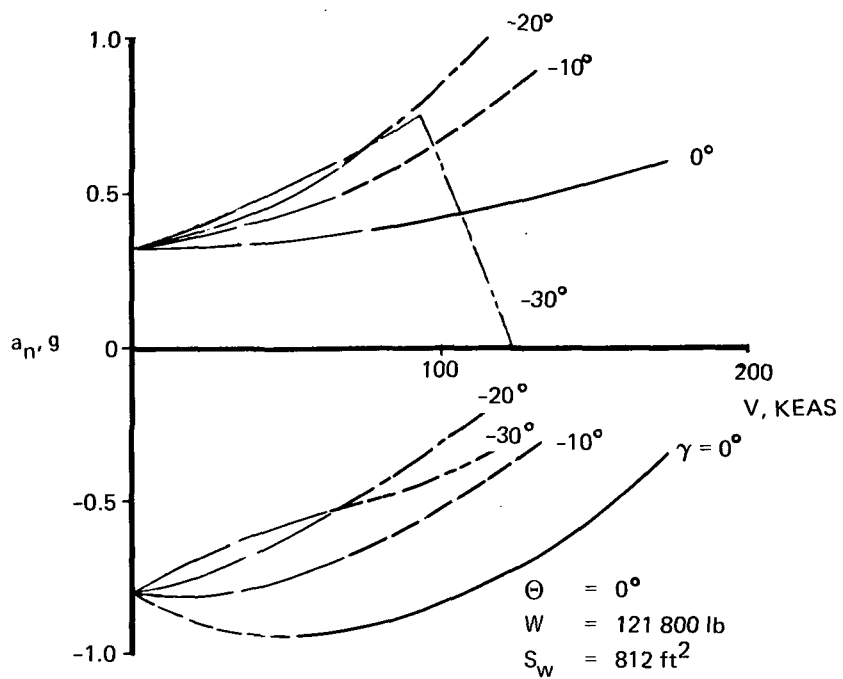
FIGURE 40.—POWERED MANEUVERING CAPABILITY NORMAL
TO FLIGHTPATH
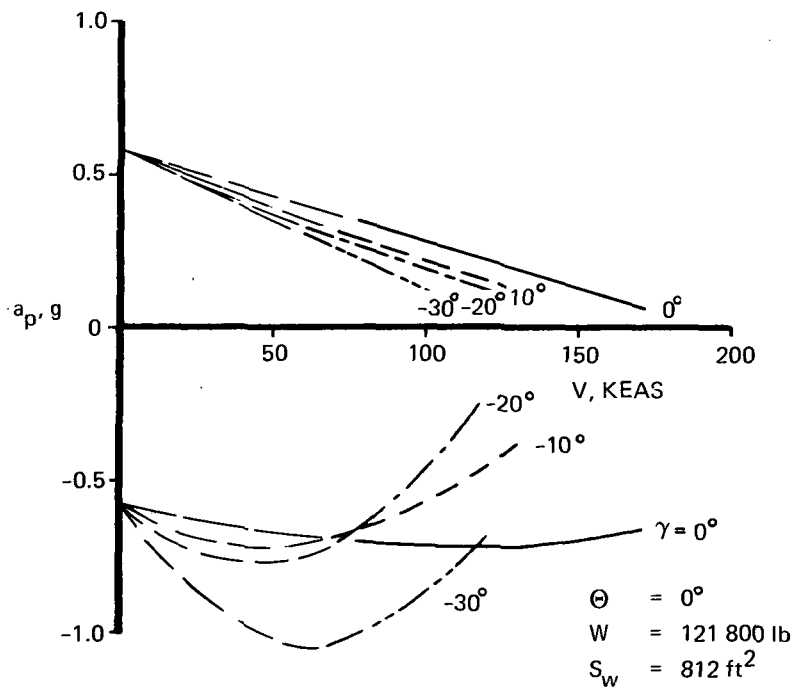
FIGURE 41.—POWERED MANEUVERING CAPABILITY PARALLEL
TO FLIGHTPATH

Maneuver Time Histories

The time histories of transition from one flight condition to another are shown in figures 42 and 43. The vehicle is assumed to be initially in level flight at 160 kn in the powered-lift mode. It is decelerated to $V = 100$ kn in level flight; a maneuver follows that ends at a specified speed and flightpath angle. Two such maneuvers are shown (figs. 42 and 43). In the first, the deceleration to 100 kn is followed by a constant-speed maneuver to a -11° glide slope (rate of sink = 2000 ft/min). This glide slope is maintained while the speed is reduced to approximately 28 kn; then at this speed, the glide slope is steepened to -20° and the vehicle is on a stabilized flightpath at a sink speed of 1000 ft/min, which will continue until flare and touchdown. The rather involved thrust and thrust vector angle settings used to follow this trajectory are shown; the initial deceleration begins at 10 sec, and it will be noted that the various fan thrusts are initially different to trim the aircraft.

Figure 43 shows a profile to achieve the same end point, but with one continuous deceleration. From 160 kn the aircraft is slowed to 100 kn, and at this point it begins to descend. The final condition of sink speed = 1000 fpm, $\gamma = -20°$, is achieved 18 sec after the start of the maneuver, slightly faster than in the previous case. This procedure is clearly simpler, but fuel consumption is slightly (approximately 5%) higher. As before, the final flight condition would continue until flare.

Of note is the fact that considerable thrust margin exists at all times during these maneuvers, the worst point being at $t = 32$ sec on the first maneuver. This indicates that most trajectories of practical interest are within the capabilities of this aircraft.

Two touchdown profiles are shown in figure 44, the thrust and thrust vector angle modulations required to perform these maneuvers are shown in figures 45 and 46. It will be noted that the upper, steeper trajectory takes more time and fuel and requires higher thrust levels just prior to touchdown. In neither case are the thrusts so high that the vehicle's maneuvering capability is impaired.

A useful region for maneuver and descent at speeds below 140 kn exists at approach slope angles up to -24°. At very low speeds ($V \rightarrow 0$) no limit on approach angle exists.
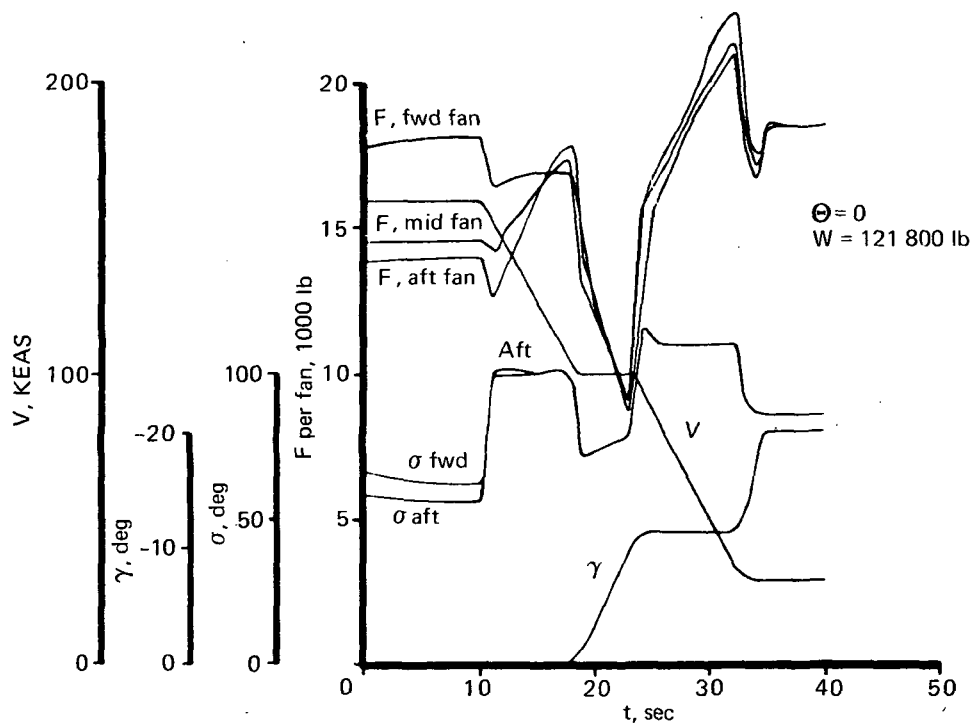
FIGURE 42.—DESCENT PROFILE; TWO-STAGE DECELERATION
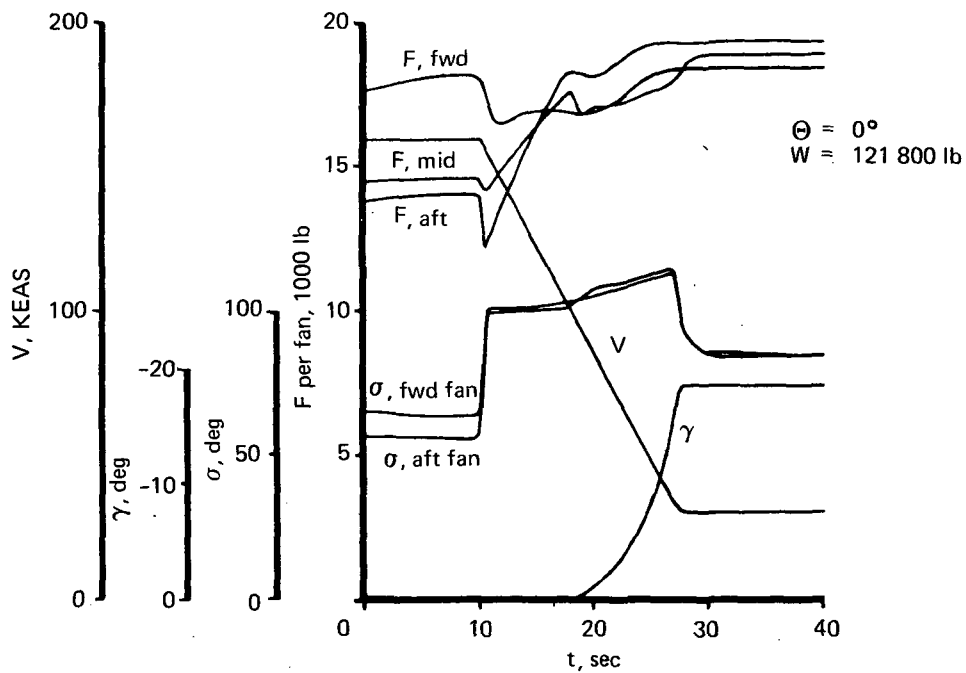
FIGURE 43.—DESCENT PROFILE; ONE-STAGE DECELERATION
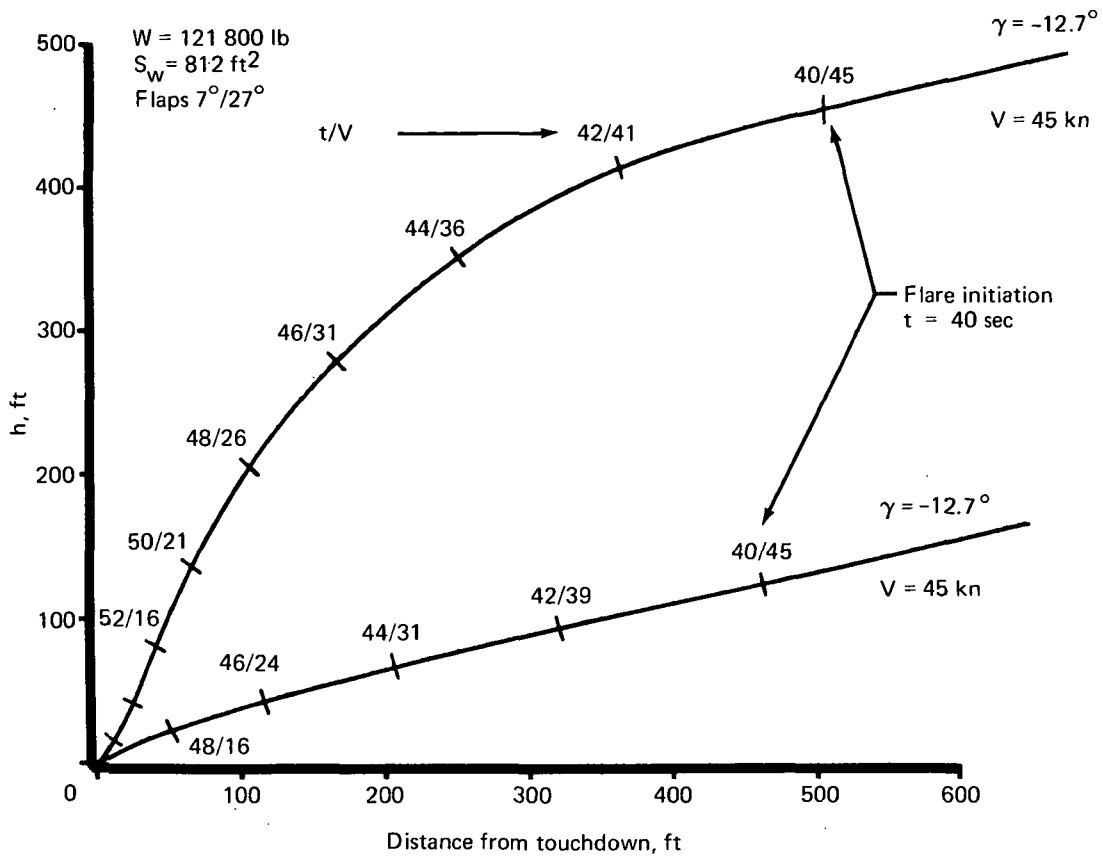
FIGURE 44.—TWO FLARE TRAJECTORIES
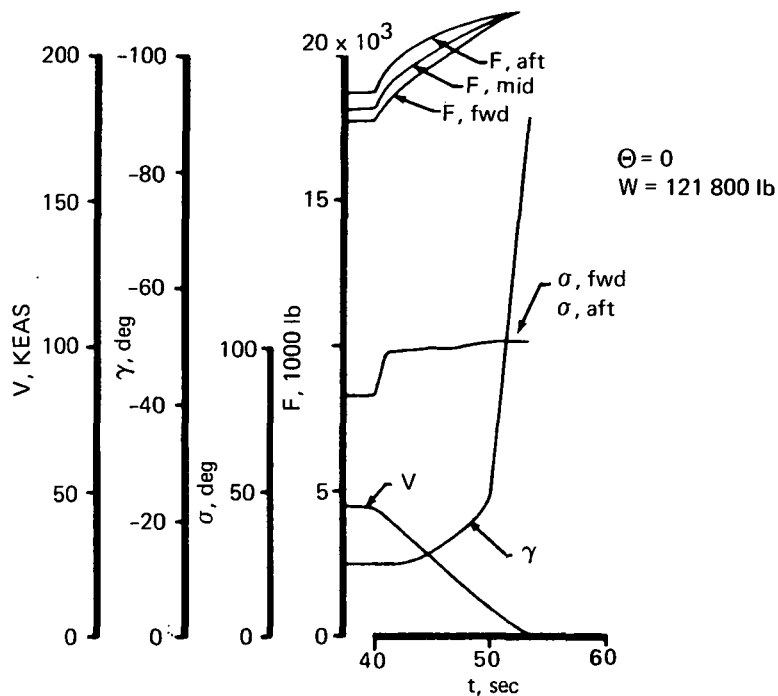
FIGURE 45.—CONSTANT γ FLARE PROFILE
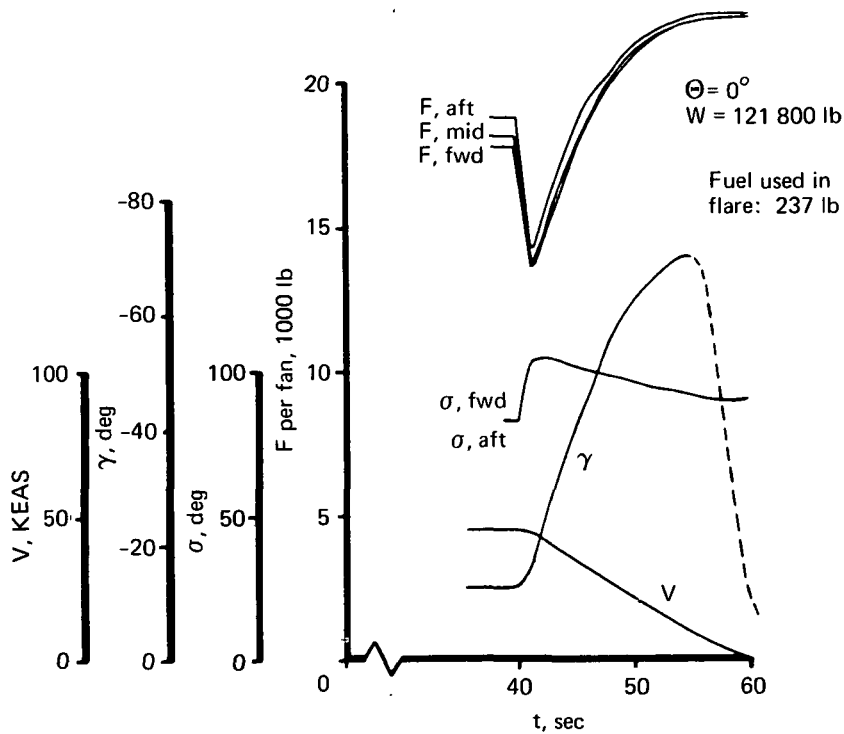
FIGURE 46.—CIRCULAR ARC FLARE PROFILE

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NOISE

The study guidelines set a perceived noise level goal for these airplanes of 95 PNdB at the 500-ft sideline. With this goal in mind, the actual noise output at the 500-ft sideline and the total area subjected to noise levels greater than the guideline level are presented.

INSTALLED NOISE CHARACTERISTICS

The lift fans with a design pressure ratio of 1.25 have a basic noise level, with 20 000 lb thrust at 500 ft, of 91.2 PNdB. An increment in perceived noise level of 2.8 dB is identified in the "Design Guidelines" section for a pressure ratio of 1.35. The noise scaling with thrust level is derived from the relationship that noise in dB is proportional to the power at the source. Putting power in terms of thrust, the noise increment becomes proportional to 10 times the log of the thrust ratio.

$$\Delta \text{PNdB} = 10 \log F_2/F_1 \tag{6}$$

and the noise increment for more than one engine

$$\Delta \text{PNdB} = 5 \left(1 + \sqrt{\sin\beta}\right) \log N \tag{7}$$

where β is the elevation angle between the noise source and the sideline listening point. At an altitude of 180 ft, this angle is 20° to the 500-ft sideline. N is the number of engines.

The noise at the 500-ft sideline of the six-engine baseline airplane with 100 passengers, the scaled airplane with 150 passengers, and the four-engine configuration (100 passengers) are listed on table 2. It is apparent that the 100-passenger airplanes have almost identical noise outputs and that the noise of the 150-passenger airplane is greater in proportion to the log of the gross weights or thrust required.

CABIN NOISE

The flight crew and passenger compartments for these V/STOL aircraft were all designed for speech interference levels of 75 dB during takeoff and 70 dB during cruise. This design requirement is the same as that used for current commercial aircraft.

TABLE 2.—NOISE SUMMARY AT 500-FT SIDELINE

	6 engines		4 engines
	100 passengers	150 passengers	100 passengers
Base: F = 20 000 lb, R_F = 1.25 One engine = PNdB Fan treatment—splitters/length	91.2 4/19 in.		91.4 5/11 in.
Noise increments ΔPNdB R_F to 1.35 N engines Thrust level	2.8 6.1 0.4	2.8 6.1 2.5	2.8 4.8 1.3
Total at 500-ft sideline	100.5	102.6	100.3

Fiberglass batt acoustical/thermal insulation is used on each of these aircraft. The weight of insulation material is a function of passenger cabin surface area. This relationship is based on the insulation and soundproofing used on the 707, 727, and 737 aircraft and is modified to account for the engines over the cabin.

The total insulation installed is set by the vertical takeoff since all engines operate at this time. In cruise, the cabin noise level will be lower than required since only two high bypass ratio engines are used.

COMMUNITY NOISE

The noise characteristics for the V/STOL transports are presented in terms of noise contours and airport/community interface. The takeoff noise contours and associated flightpath for the six-engine 100-passenger airplane are shown in figure 47. The flight profile consists of a vertical takeoff and transition followed by climb. An altitude of 500 ft is reached at a distance of 1000 ft. At 4000 ft from liftoff, the altitude is 1000 ft. The 95 PNdB contour encloses an area of 69 acres (about 0.11 miles^2). A 100-PNdB contour surrounds an area of about 31.5 acres.

The landing noise contours and the approach path for this airplane are shown in figure 48. The approach is made at part throttle down to an altitude of 500 ft, 1000 ft from touchdown. As the descent continues, the thrust is increased to decelerate and flare. The resulting 95 PNdB and 100 PNdB contours enclose 73 acres and 35 acres, respectively.

The area subjected to noise levels greater than 95 PNdB during takeoff and landing maneuvers of the V/STOL transport is more than 100 times smaller than that of conventional short-haul

TAKEOFF PROFILE

NOISE CONTOURS
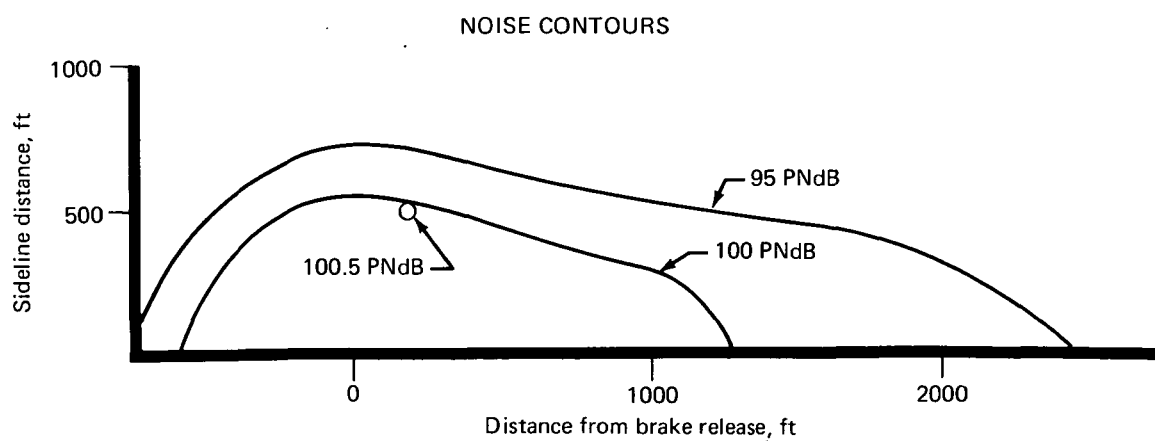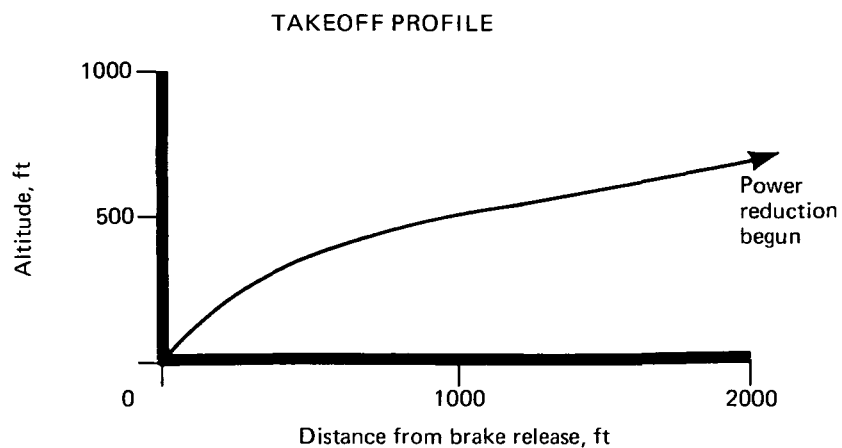
Area enclosed within 95 PNdB = 69   acres (0.11 mile^2)
100 PNdB = 31.5 acres

FIGURE 47.—TAKEOFF NOISE—SIX ENGINES, 100 PASSENGERS

LANDING PROFILE

F_n

Alt

F_n—net thrust/engine, lb

Altitude (ft)

20,000

10,000

0

1000

500

0

1000

2000

Distance, ft

NOISE CONTOURS

Sideline distance, ft

1000

500

0

0

1000

2000

95 PNdB

99.5 PNdB

100 PNdB

Distance, ft
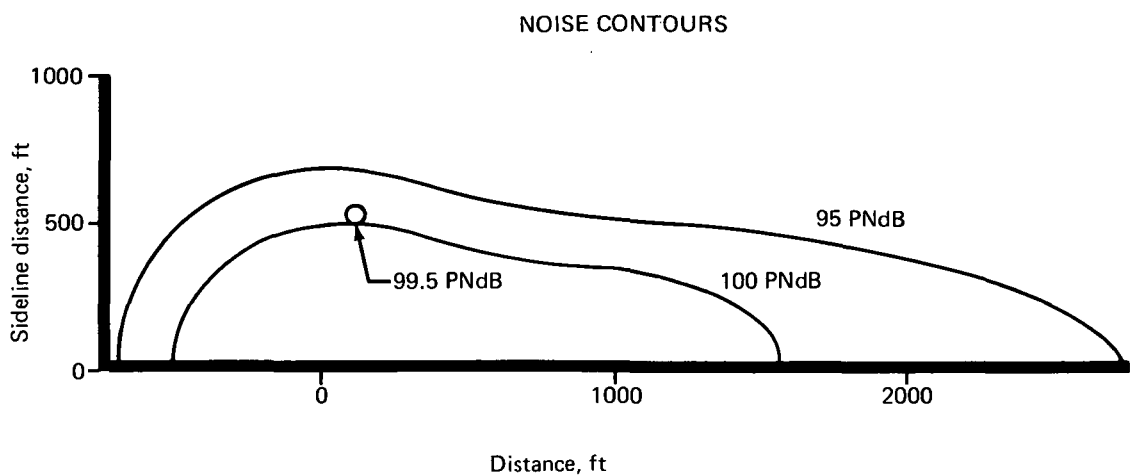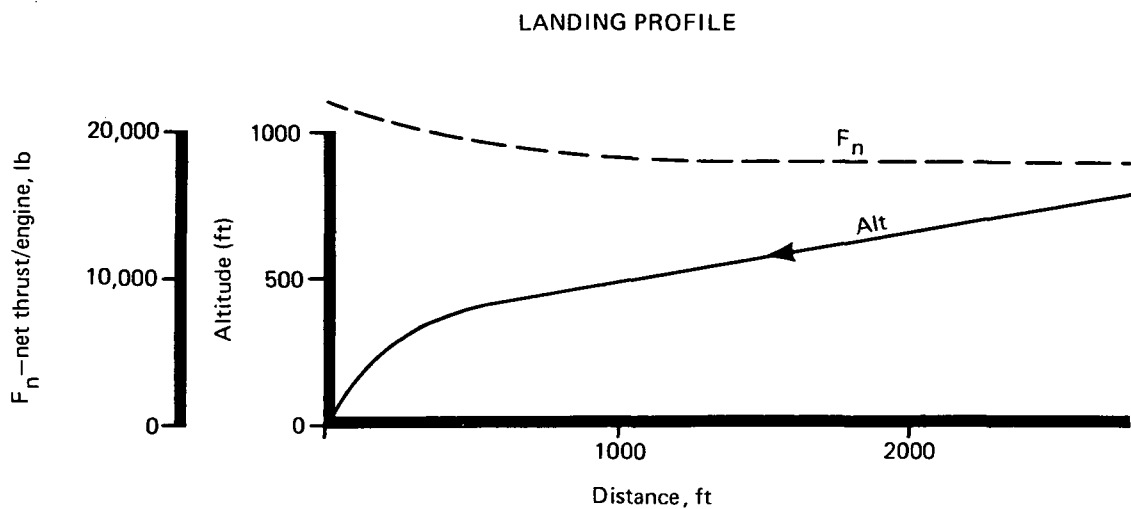
Area enclosed within 95 PNdB = 73 acres
100 PNdB = 35 acres

FIGURE 48.—LANDING NOISE—SIX ENGINES, 100 PASSENGERS

transports. Although the noise goal of 95 PNdB at 500 ft has not quite been met, the introduction of V/STOL transports will not have a disturbing effect on the adjacent community.

The impact of the V/STOL transport on the community may be further assessed by comparing the takeoff and landing noise contours with the interface between the airport and the adjacent community. The airport/community interface was determined by analyzing a large number of STOL and short-haul airports. The airport/community interface is shown in figure 49. The curves are labeled to show the percentage of the airport/community boundaries that are outside a given contour. For example, take the 96% contour: 96% of the airports have the runway axis/community interface more than 500 ft from the runway center.

The superposition of the 95-PNdB vertical takeoff and landing footprints on the community interface contours is also shown in figure 49. The noise footprint lies within the airport/community boundaries of 96% of the short-haul airports. This again indicates a good situation for V/STOL transports.

FLIGHT PROFILES

Altitude, ft

2000

1000

-10   -5   0   5   10

Distance, 1000 ft

Community boundaries
(% of medium haul airports)

96
88
76
60

95 PNdB
contours

5

-15   -10   -5   5   10   15

Distance, 1000 ft

5
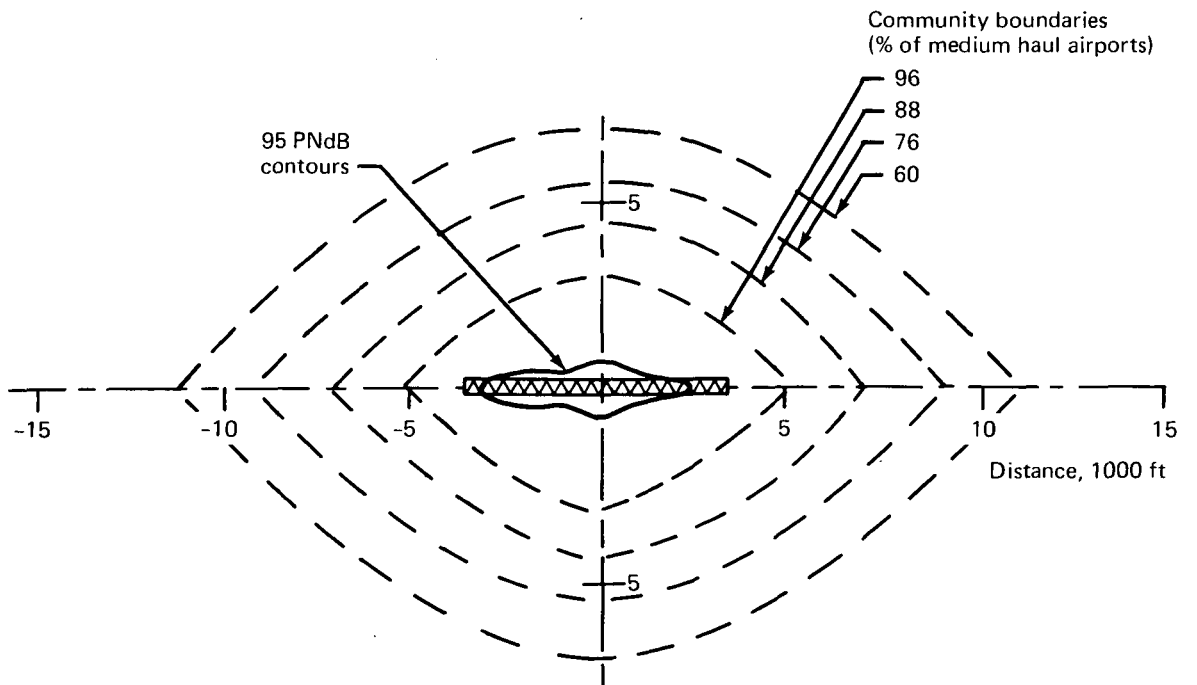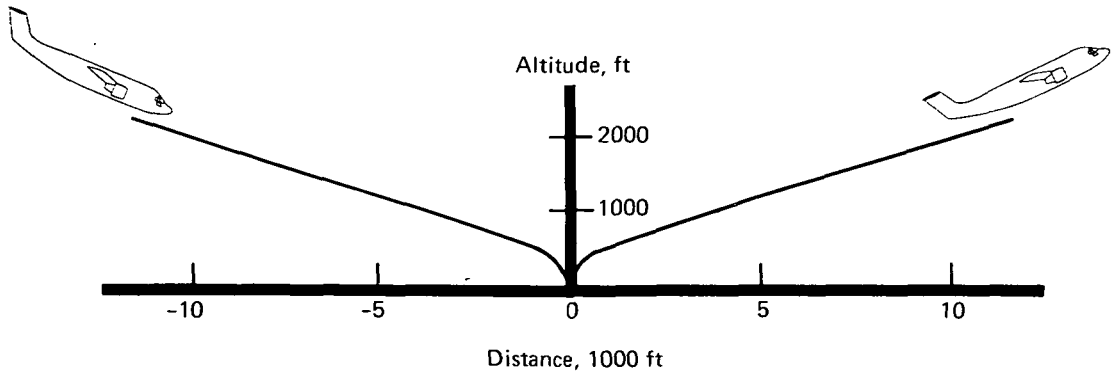
FIGURE 49.—AIRPORT/COMMUNITY BOUNDARY AND AIRPLANE NOISE

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ECONOMICS

The economics of the 1985 V/STOL transports has been estimated in terms of the manufacturing cost or initial investment and the direct operating cost.

MANUFACTURING COST

The manufacturing costs of each airplane were estimated at airframe cost rates of $90 and $110 per pound. The effect on the baseline airplane of varying the engine cost by ±20% was also found. The effect of design range, fan failure philosophy, number of engines, and number of passengers may be seen by comparing the study airplanes. The costs are tabulated in table 3.

TABLE 3.—MANUFACTURING COST COMPARISONS

Fan failures	No. of passengers	No. of fans	VTO design range (nmi)	Airframe at $90/lb	$110/lb
				Cost ($ million)	
Safe fan failure	100	6	400	(12.5) 11.5 (10.4)	12.9
No fan failure	100	6	400	10.5	11.8
No fan failure	100	6	200	9.8	11.0
No fan failure	100	4	200	9.5	10.7
Safe fan failure	150	6	400	16.0	18.0

(±20% on engine cost)

The costs relative to that of the baseline airplane are shown in figure 50. The effect of the design variables on cost is easily seen. The design for no fan failure reduces costs about 9%, and the four-engine design costs are about 3% less than the equivalent six-engine airplane.

DIRECT OPERATING COST

The direct operating costs were estimated for each design to show the effects of the design parameters. In addition, the sensitivity of DOC to utilization, manufacturing cost, and operation were found.
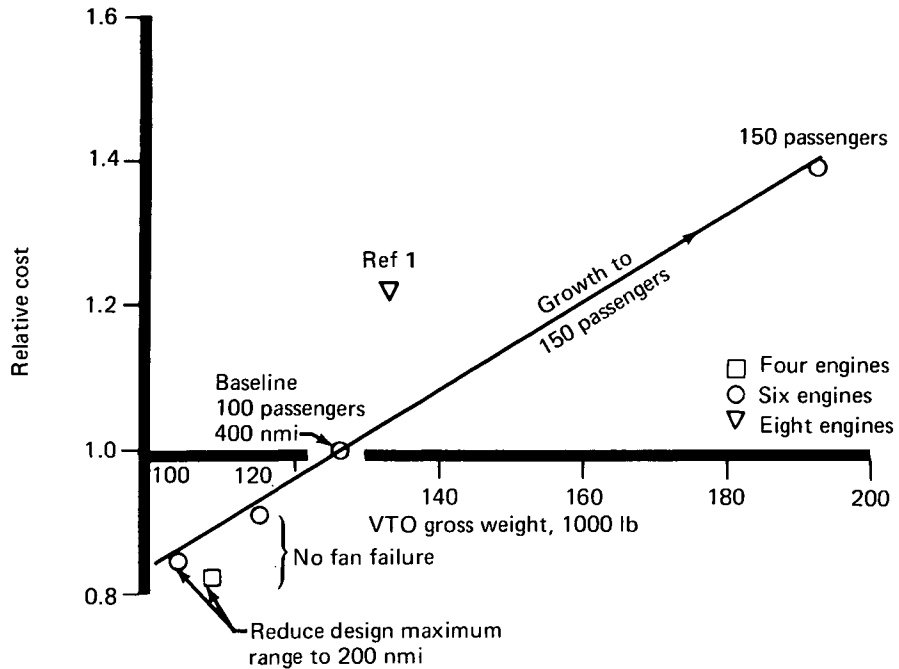
FIGURE 50.—RELATIVE REMOTE FAN V/STOL TRANSPORT COSTS

The variation of DOC for each design as a function of installed thrust/weight ratio is shown in figure 51 at an operational range of 200 nmi. The effect of fan failure philosophy causes an 8% change in DOC for a 5% change in F/W. Increasing the passenger capacity by 50% and the F/W by 1% reduces DOC by 10%. Once again, the four-engine design has an advantage over the six-engine airplane; the four-engine DOC is 7% lower. The F/W reduction shown is for the fan design. The gas generators are oversized, with a potential F/W of 1.49, to handle the case of a gas generator failure.

The influence on DOC of initial cost and utilization is shown in table 4 for the baseline airplane at the VTO design range of 400 nmi.

The effect of operating range is shown in figure 52. The shape of the curves is independent of configuration. The increments due to design differences are the same as those shown in figure 51. The range of DOC for conventional short-haul transports from reference 1 is shown for comparison. The V/STOL transport DOCs are from 30% to 100% higher than conventional aircraft at ranges of 200 to 300 nmi.
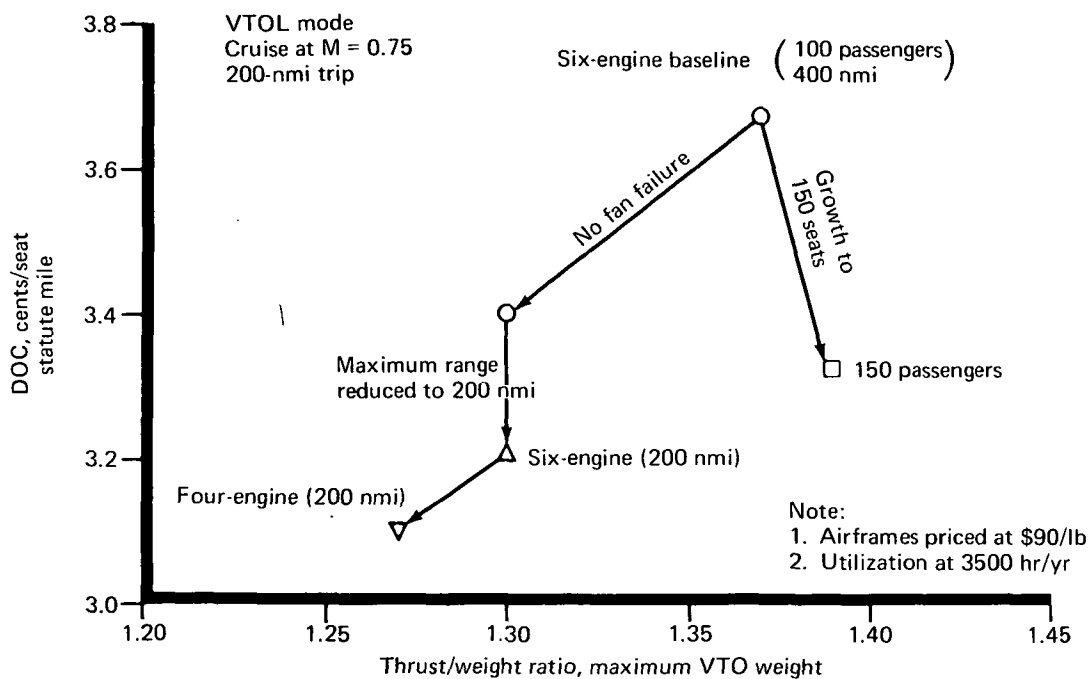
FIGURE 51.—DIRECT OPERATING COST—EFFECT OF STUDY PARAMETERS

TABLE 4.—DIRECT OPERATING COST SENSITIVITY

Baseline airplane 984-139 Range—400 nmi		
Manufacturing cost ($ millions)	Direct operating cost (cents per available seat statute mile) at utilization rates	
	2500 hr/yr	3500 hr/yr
11.5	3.00	2.60
12.9	3.20	2.75

FIGURE 52.—DIRECT OPERATING COST

CONCLUSIONS

These studies led to conclusions about weight, cost, and operation. The differences due to fan failure philosophy are most important.

1. The development of an "operationally reliable" fan, one that will not fail in flight, will allow design of a V/STOL transport with four fans. The four-fan airplane is slightly heavier than the six, 107 900 lb to 103 600 lb. This is due, in part, to the higher cruise thrust loading, shorter reaction control moment arms, and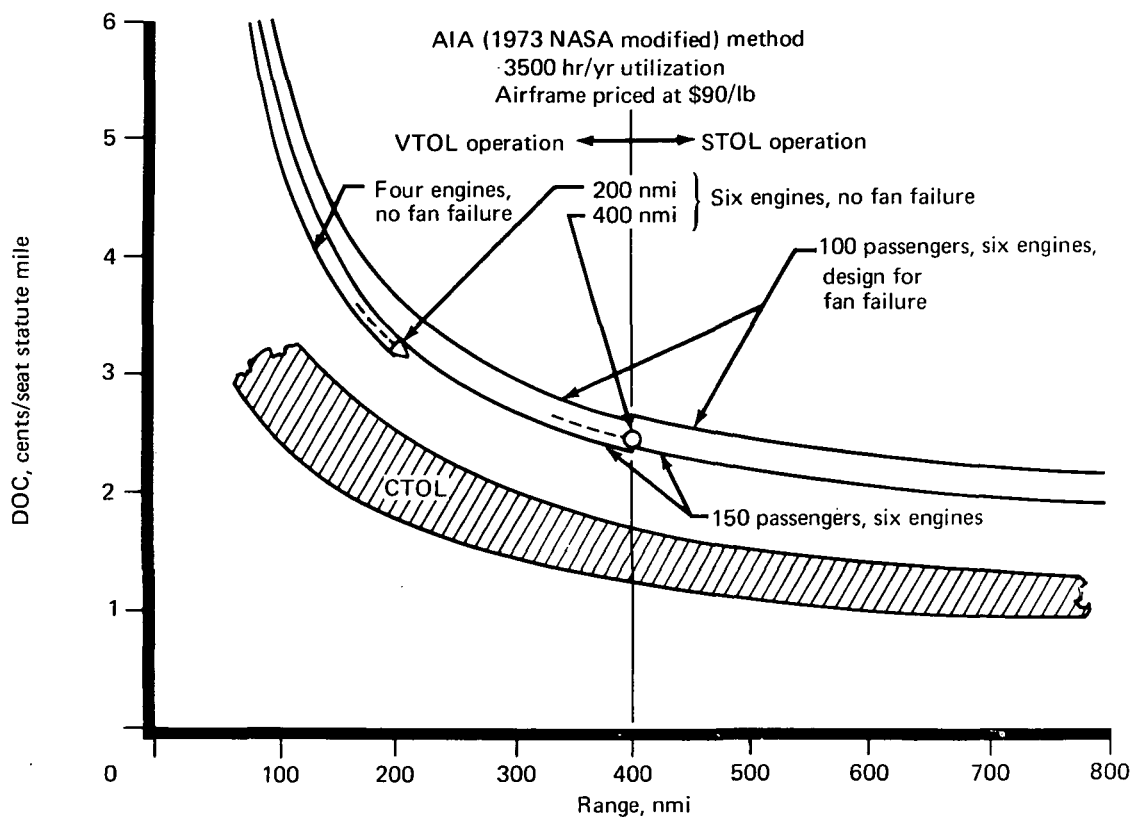 lighter wing loading. In spite of this, the system can be attractively integrated, and the resulting cost and DOC are about 8% lower than that of the six-engine airplane with equal performance.

2. Cruise at M = 0.85, instead of M = 0.75 can be achieved at about an 8% increase in operating empty weight. Design changes to achieve M = 0.85 include sweeping the wing, refairing the cockpit, and refairing the junctures of fuselage, wing, and tail.

    Cruise at M = 0.65 will only save about 1% of the mission fuel. Redesign from M = 0.75 to optimize at M = 0.65 included increasing the wing thickness and using a noncritical airfoil section. The full benefit of speed reduction can be achieved by operating the basic, M = 0.75, airplane at M = 0.65 without redesign.

3. A useful steep approach corridor is available. Maneuver and descent can be accomplished at approach slopes down to -24°.

4. The 95-PNdB noise contour of these V/STOL transports encloses an area more than 100 times smaller than that of current short-haul jet aircraft. The combined takeoff and landing 95-PNdB noise contour will be within 96% of the short-haul airport/community boundaries. The introduction of V/STOL transports will have a salutary effect on the adjacent community.

5. Initial cost of V/STOL transports will be about twice that of conventional transports. The four-engine design, with "operationally reliable" fans, will be the best.

    Direct operating costs of the V/STOL transports, on a 200- to 300-nmi mission will be 30% to 100% higher than conventional airplanes.

Boeing Commercial Airplane Company
  P. O. Box 3707
    Seattle, Washington 98124, March 25, 1974

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APPENDIX A

EXCERPTS FROM STUDY GUIDELINES AND DESIGN CRITERIA
FOR
CONCEPTUAL DESIGN OF V/STOL LIFT-FAN TRANSPORTS

FLIGHT SAFETY AND OPERATING CRITERIA

Handling Qualities Criteria (Low-Speed Powered-Lift Mode)

Except where specific criteria are given, handling qualities shall comply with the recommendations of AGARD-R-577-70. Where possible, two levels of criteria are stated; the first is intended for normal operation and the second for operation following any reasonable single failure of the powerplant or control system. Definitions of the two levels are as follows:

- Level 1: Flying qualities are as near optimal as possible and the aircraft can be flown by the average commercial pilot.

- Level 2: Flying qualities are adequate to continue flight and land. The pilot workload is increased but is still within the capabilities of the average commercial pilot.

Attitude Control Power (SL, ISA + 31° F)

- Level 1: At all aircraft weights and at all speeds up to V_{con}, the low-speed control power shall be sufficient to satisfy the most critical of the two following sets of conditions:

a) First conditions—to be satisfied simultaneously

1) Trim with the most critical cg position.

2) In each control channel provide control power, for maneuver only, equal to the most critical of the requirements given in table A1.

These maneuver control powers are applied so that 100% of the most critical and 30% of each of the remaining two need occur simultaneously.

TABLE A1.—MANEUVER CONTROL REQUIREMENTS—LEVEL 1

Axis	Maximum angular[a] acceleration after a step input, rad/sec^2		Attitude angle in 1 sec after a step input, deg	
	VTOL	STOL	VTOL	STOL
Roll	± 0.6	± 0.4	±10	±6
Pitch	± 0.33	± 0.3	± 6	±5
Yaw	± 0.25	± 0.2	± 5	±3

[a]For purposes of the design study these should be construed as control moment/inertia rather than acceleration measured with a control input.

b) Second conditions—to be satisfied simultaneously

1) Trim in a 25-kn TAS crosswind with the most critical cg position.

2) In each control channel provide control power, for maneuvering only, equal to 50% of the values given in the previous table. Simultaneous control power need be no greater than 100%, 30%, 30%.

● Level 2: At all aircraft weights and at any speed up to V_{con}, the low-speed control power shall be sufficient to satisfy, simultaneously, the following:

a) Trim after any reasonable single failure of powerplant or control system.

b) In each control channel, provide control power, for maneuver only, equal to the most critical of the requirements given in table A2. Simultaneous maneuver control power need be no greater than 100%, 30%, 30%.

TABLE A2.—MANEUVER CONTROL REQUIREMENTS—LEVEL 2

Axis	Maximum angular[a] acceleration after a step input, rad/sec^2		Attitude angle in 1 sec after a step input, deg	
	VTOL	STOL	VTOL	STOL
Roll	±0.3	±0.2	±5	±3
Pitch	±0.2	±0.2	±3	±3
Yaw	±0.15	± 0.15	± 2	±2

[a]For purposes of the design study these should be construed as control moment/inertia rather than acceleration measured with a control input.

Flightpath Control Power (SL to 1000 ft, ISA + 31° F)

VTOL (0-40 kn TAS and zero rate of descent).[†]—At all aircraft weights and at the conditions for 50% of the maximum attitude control power specified in the preceding paragraphs, it shall be possible to produce the following incremental accelerations for height control.

- Level 1:

  a)   In free air, ±0.1 g

  b)   With wheels just clear of the ground, -0.10 g, +0.05 g

- Level 2:

  a)   In free air, -0.1 g, +0.05 g[†]

  b)   With wheels just clear of the ground, -0.10 g, +0.00 g

It shall also be possible to produce the following horizontal incremental acceleration, but not simultaneously with height control.

- Level 1: ±0.15 g

- Level 2: ±0.10 g

At all aircraft weights it shall be possible to produce the following stabilized thrust/weight ratios without attitude control inputs.

- Level 1: F/W = 1.05 in free air

- Level 2: F/W = 1.03 in free air

VTOL and STOL approach (40 kn to V_{con}).—At the maximum landing weight and in 25-kn crosswind, the aircraft shall be capable of making an approach at 2000 fpm rate of descent while simultaneously decelerating at 0.15 g along the flightpath.

---

†This condition is critical in establishing the total installed power for these airplanes.

It shall be possible to produce the following incremental normal accelerations in less than 1.5 sec for flightpath tracking when more than 0.1 g but less than 0.3 g can be developed by aircraft rotation using pitch control.

- Level 1: ±0.1 g

- Level 2: ±0.05 g

It shall be possible to produce the following incremental normal acceleration in less than 0.5 sec for flare and touchdown control when more than 0.1 g but less than 0.15 g can be developed by aircraft rotation using pitch control.

- Level 1: ±0.1 g

- Level 2: ±0.05 g

VTOL Control System Lags (SL to 1000 ft, ISA + 31° F)

The effective time constant (time to 63% of the final value) for attitude control moments and for flightpath control forces shall not exceed the levels given in table A3.

The step input is assumed to be applied at the pilot's control.

TABLE A3.—FLIGHTPATH CONTROL FORCE EFFECTIVE TIME CONSTANTS

Item	Level 1	Level 2
Attitude control moments	0.2 sec	0.3 sec
Flightpath control forces	0.3 sec	0.5 sec

VTOL Takeoff and Landing Safety Criteria

With the selected takeoff or landing operational procedure, any reasonable single failure of the powerplant or control system, together with a simultaneous discrete gust about any axis, as defined by MIL-F-8785B (ASG), the aircraft shall be capable of continued sustained flight.

The airfield shall be assumed to be at sea level and the atmosphere ISA + 31° F with a 25-kn crosswind.

STOL Takeoff and Landing Safety Criteria

A requirement comparable to that given above for VTOL shall be satisfied.

The following relationship between the various ground speeds shall hold:

$$V_{LOF} \geqslant V_R \geqslant V_1 \geqslant 1.05 \, (V_{MCG} \text{ and } V_{MCA})$$

where

V_{MCG} = minimum control speed on the ground, FAR XX 149

V_{MCA} = minimum control speed in the air, FAR XX 149

V_1 = critical decision speed, FAR XX 53

V_R = rotation speed

V_{LOF} = liftoff speed, FAR XX 53

The obstacle clearance speed V_2 shall satisfy the following relationships

V_2 $\geqslant V_{LOF}$

$\geqslant 1.15 \, V_{MCA}$

$\geqslant V_{MCA} + 10 \, kn$

$\geqslant 1.2 \, V_{min}$

V_{min} = minimum flying speed with gear down, FAR XX 49.

The angle of attack during climbout shall be 10° or more below the angle of attack for stall, in the takeoff configuration, with gear down, and the most critical powerplant failure.

The climbout gradient, in the takeoff configuration with gear down and powerplant fully operative, and with gear up and the most critical powerplant failure, shall be at least 6.7% (15:1).

The takeoff field length shall be the greatest of

a)   115% of all-engine takeoff distance to 35 ft

b)   100% of the critical powerplant failure takeoff distance to 35 ft

c)   100% of the accelerate stop distance

The approach speed at the 35-ft threshold, V_{AP}, shall satisfy the following relationships

$$V_{AP} \geq 1.15\, V_{MCA}$$

$$\geq V_{MCA} + 10 \text{ kn}$$

$$\geq 1.2\, V_{min}$$

and

$$\alpha_{AP} \leq \alpha_{stall} - 10°$$

The landing climbout gradient at V_{AP} under the following conditions shall be at least 3.33% (30:1).

a)   Powerplant at full power

b)   Gear down

c)   Landing flap angle

or

a)   The most critical powerplant failure with the remaining powerplant at full power

b)   Gear up

c)   Landing flap angle

The landing climbout gradients under the above conditions, but with a configuration change, shall be at least 6.7% (15:1). The landing field length is defined as the total distance from the 35-ft threshold, divided by 0.7.

Conversion Requirements (STOL and VTOL)

It must be possible to stop and reverse the conversion procedure quickly and safely without unduly complicated operation of the powered-lift controls.

The maximum speed in the powered-lift configuration shall be at least 30% greater than the power-off stall speed in the converted configuration for level 1 operation; the speed in the powered-lift configuration shall be at least 10% greater than the power-off stall speed for the level 2 operation.

Fuel Reserves

The fuel reserves listed in table A4 are to be calculated on the basis that the flight to the alternate airport shall be at the most economic fuel consumption condition, and the hold at 5000 ft is on the descent at the alternate airport.

TABLE A4.—FUEL RESERVES

Item	VTOL	STOL
Holding at 5000 ft and most economical speed	20 min	30 min
Flight to alternate airport	50 nmi	100 nmi

A V/STOL aircraft on a STOL mission may be able to land vertically, if necessary, at the destination. The aircraft may, in this case, use the VTOL reserves given in the table.

PERFORMANCE

Pilot and Aircraft Operating Capability

Table A5 lists pilot and aircraft operating capabilities. Airports are at sea level with an ambient air temperature of 90° F.

Payload-Range

Each passenger will be assumed to have a weight of 200 lb (160 lb per passenger and 40 lb of nonrevenue baggage). No revenue cargo is assumed. Nonstop range in standard atmosphere, still air, at maximum payload, at normal cruise airspeed, and without using reserve fuel shall be as dictated in the statement of work.

TABLE A5.—PILOT AND AIRPORT OPERATING CAPABILITIES

Operating function	Assumed capability
Macimum deceleration on the ground	0.4g
Rolling coefficient of friction	0.03
Pilot reaction time to initiate any emergency procedure, excluding the response time of any mechanism activated	2 sec
Time lag after touchdown for activation of lift spoiling and decelerating devices, excluding the response time of any mechanism activated	0.5 sec for automatic
	1.0 sec for nonautomatic
Maximum rate of descent at 35 ft altitude	800 fpm
Maximum rate of descent at touchdown for performance calculation	300 fpm
Maximum rate of descent for gear design	600 fpm

Cruise Airspeed and Altitude

Cruise speed shall be as specified in the statement of work.* Cruise altitude shall be such that the cruise distance is at least one-half of the total stage length.

Mission Profile

The mission profile is shown in table A6. Maximum airspeed shall not exceed 250 kn IAS below 10 000 ft altitude.

STOLport Definition

The STOLport has a length of 1500 ft with 100-ft extensions at each end. The aircraft may start its ground roll from the runway extension.
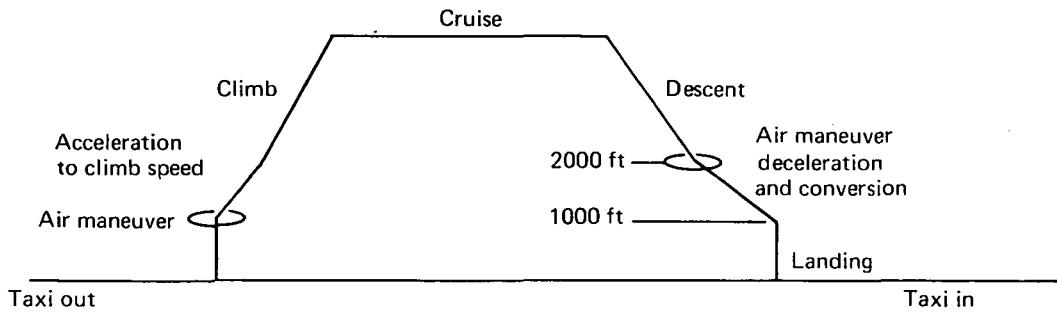
NOISE LEVELS

The noise level goal is 95 PNdB on a 500-ft sideline.

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*Baseline design condition is $M = 0.75$ or 350 kn whichever is less.

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TABLE A6.—V/STOL MISSION PROFILE DEFINITION

Segment	Time		Distance		Remarks
	VTOL	STOL	VTOL	STOL	
Taxi out	1 min	2 min	0	0	
Takeoff, transition, and conversion to conventional flight	0.5 min	0.5 min	0	0	
Air maneuver (origin)	0.5 min		0	0	
Acceleration to climb speed	As calculated				
Climb	As calculated				At optimum climb speed
Cruise	As calculated				At constant integral 1000-ft altitudes (no en route altitude changes)
Descent to 2000 ft	As calculated				5000 fpm maximum rate of descent
Air maneuver at 2000 ft (destination)	1.5 min	3 min	0	0	
Decelerating approach and conversion to powered lift flight, 2000 ft to 1000 ft	As calculated		0	0	1000 fpm maximum rate of descent
Transition and landing from 1000 ft to touchdown	As calculated		0	0	1000 fpm maximum rate of descent down to 35 ft; 600 fpm maximum rate of descent below 35 ft
Taxi in	1 min	2 min	0	0	

GENERAL DESIGN GUIDELINES

Number of Crew and Cabin Attendants

Accommodation and equipment shall be provided for a flight crew of two and for one cabin attendant per 50 passengers. In addition, some provision shall be made on the flight deck for an occasional flight observer. Each crewman plus gear weighs 190 lb, and each cabin attendant plus gear weighs 140 lb.

Aircraft Design Life

Aircraft design life shall be 40 000 hr.

Baggage Hold

The baggage hold shall have sufficient volume to store at least 5.0 ft^3 of baggage per passenger.

Auxiliary Power Unit (APU)

The aircraft shall be equipped with an APU to meet the needs of starting, ground air conditioning, and heating.

Aircraft Materials

The aircraft designs are to be based on 1985 operation.

All-Weather Capability

It is to be assumed that, by 1985, a system to permit all-weather operation will have been established and that the V/STOL short-haul transport system will use it.

Center-of-Gravity Limits

The allowable center-of-gravity travel shall provide trim sufficient to offset a payload shift of ±5% of the passenger cabin length.

Cruise Stability

The aircraft as configured for cruise flight shall be statically stable with a stability margin of 0.05 at the critical center of gravity without stability augmentation.

Standard Weight Items

The weights shall be as provided in table A7.

TABLE A7.—STANDARD WEIGHT ITEMS

Item	Weight
Wheels, tires, and brakes	1.25% TOGW
Instruments (flight and navigation) Electrical (excluding generating equipment) Electronics (communication, flight, and navigation) Auxiliary power unit installation	1200 lb
Seats and belts Passenger: Double Triple Crew seats: Cabin crew Flight crew	 23 lb/passenger 22 lb/passenger 16 lb/crew member 55 lb/crew member
Lavatory	300 lb/unit
Galley (predicated on 100 passengers) Food and beverage	5 lb/passenger 4 lb/passenger

APPENDIX B

PROPULSION SYSTEM CHARACTERISTICS

A family of lift/cruise fans for 1985 operational V/STOL commercial transports was developed for use during this conceptual design study of such aircraft. The fans are based on General Electric lift fans, and the size, weight, and scaling rules are taken from the General Electric study (references 3 and 4).

The fans are tip turbine powered from remote gas generators. The basic characteristics tabulated in table B1 consist of: the maximum control thrust per/lb of total airflow; the bypass ratio; the ratio of cruise and climb thrust to the maximum control static thrust; and the takeoff and cruise SFC.

TABLE B1.—LIFT/CRUISE FAN CHARACTERISTICS SUMMARY

Sea level static				M = 0.75, 20 000 ft				M = 0.75, 30 000 ft			
R_F	F_{MC}/W_{tot} $\frac{lb}{lb/sec}$	SFC	BPR	$\frac{F_{CR}}{F_{MC}}$	SFC cruise	$\frac{F_{CL}}{F_{MC}}$	SFC climb	$\frac{F_{CR}}{F_{MC}}$	SFC cruise	$\frac{F_{CL}}{F_{MC}}$	SFC Climb
1.25	20.4	0.236	16.1	0.235	0.57	0.260	0.57	0.191	0.54	0.207	0.54
1.30	22.4	0.257	13.1	0.255	0.57	0.283	0.57	0.205	0.55	0.221	0.55
1.35	24.0	0.272	11.6	0.268	0.58	0.300	0.58	0.216	0.55	0.233	0.55

LIFT/CRUISE FAN DESIGN POINTS

Lift/cruise fan cycles with fan pressure ratios from 1.25 to 1.35 were calculated. These single-stage fans are driven by turbines at the blade tips. The tip turbine is driven by a turbojet gas generator. The maximum allowable gas temperature is determined by the permissible scroll temperature. The gas temperature limits and the thrust ratings associated with them are:

*Maximum control thrust, F_{MC}	$2060°R$
Maximum climb thrust, F_{CL}	$1930°R$
Maximum cruise thrust, F_{CR}	$1860°R$

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*This is the design point; it is an intermittent, transient condition.

An emergency rating 4% greater than the maximum control thrust is assumed available.

Gas Generator Description

The gas generator was designed to provide an exhaust temperature of 2060° R at its maximum operating condition. It has a 20:1 pressure ratio compressor which results in the relationship between the primary (gas generator) turbine inlet temperature and the duct and scroll temperature that is shown in figure B-1. For the limiting case of 2060° R scroll temperature, the primary temperature is 2755° R.
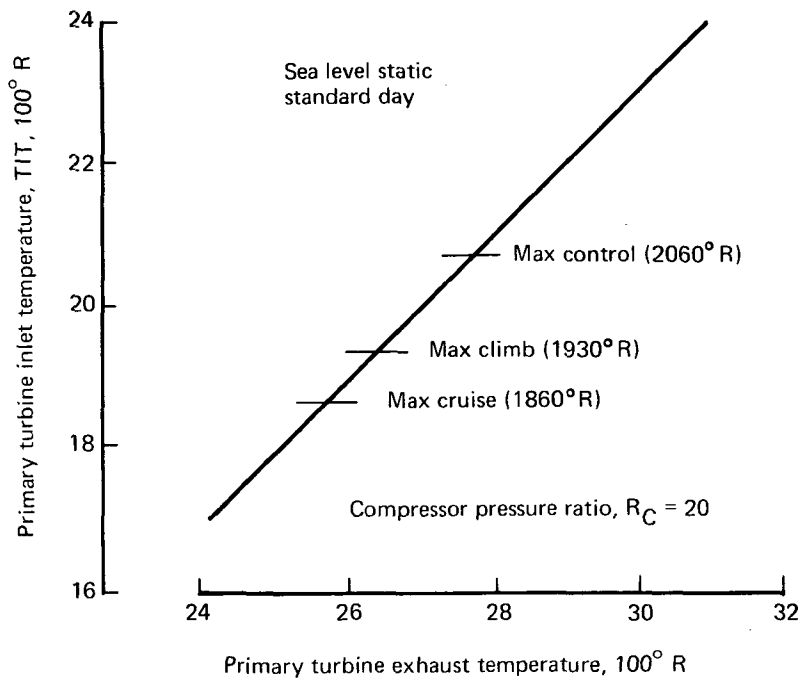
FIGURE B-1.—DESIGN POINT TURBINE TEMPERATURE

Design Point Performance

The fan performance was calculated at pressure ratios of 1.25, 1.30, and 1.35. Performance, size, and weight of three reference fan/gas generator sets were calculated for a total airflow of 1000 lb/sec at the static maximum control thrust. The low-speed performance used during V/STOL is flat rated to ISA + 31° at sea level. The value of F_{MC}, the dimensions, and the weight are listed in table B2. The dimensions are referred to in figures B-2 and B-3, which are installation drawings of the fan and gas generator. These weights and dimensions were taken from references 3 and 4.
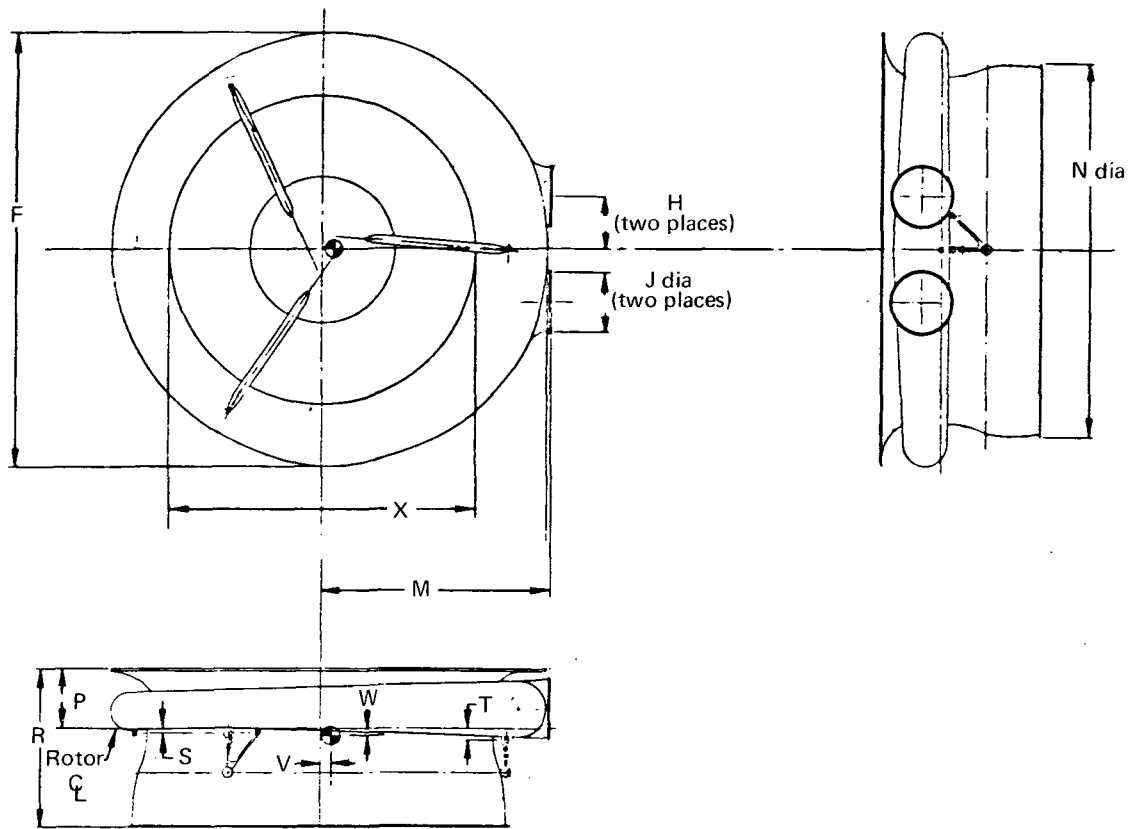
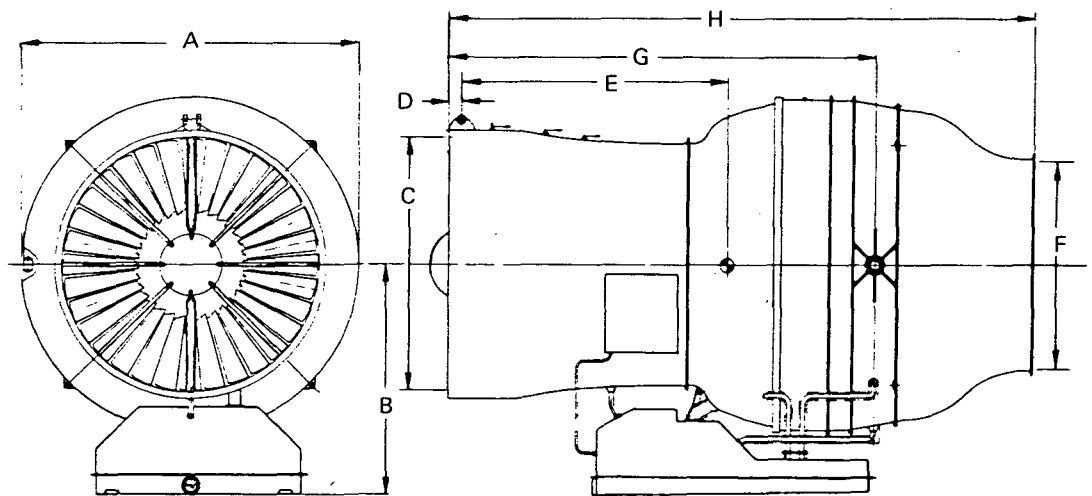FIGURE B-2.—LIFT UNIT INSTALLATION DRAWING

FIGURE B-3.—TURBOJET GAS GENERATOR INSTALLATION DRAWING

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TABLE B2.—REFERENCE DIMENSIONS AND WEIGHTS

Fan pressure ratio		1.25	1.3	1.35
Max control thrust, lb		20 400	22 400	24 000
Fan weight, lb		1 482	1 765	1 850
Gas generator weight, lb		785	912	1 025
Dimensions (figs. 1, 2), in.				
Fan	F	111.0	108.5	109.4
	H	14.5	14.3	14.4
	J	15.9	15.7	15.8
	M	59.9	58.8	59.0
	N	93.1	91.5	92.0
	P	15.4	15.2	15.2
	R	37.7	37.0	37.2
	V	2.6	2.5	2.5
	W	1.7	1.6	1.7
Fan diameter	X	79.2	77.8	77.0
Gas generator	A	33.4	36.2	38.6
	B	22.3	24.4	26.2
	C	24.9	27.2	29.0
	D	1.2	1.3	1.4
	E	26.8	29.2	31.2
	F	20.5	22.4	23.9
	G	41.4	45.1	48.2
	H	59.9	65.3	69.9

The weight and size may be scaled to other thrust levels by the thrust ratio to the appropriate power.

$$W = W_{ref} \left[\frac{F_{MC}}{F_{MC_{ref}}} \right]^{1.25}$$

$$L = L_{ref} \left[\frac{F_{MC}}{F_{MC_{ref}}} \right]^{0.5}$$

where

W   =   weight

F_{MC} =   maximum control thrust–static

L   =   any linear dimension

Fuel flow scales directly with thrust.

The complete performance of these reference systems with fan pressure ratios of 1.25, 1.30, and 1.35 is presented.

Fan Pressure Ratio 1.25–Performance

The performance (thrust and fuel flow) for the 1.25 fan pressure ratio engine is presented in figures B-4 through B-10. These data are for the reference thrust engine with F_{MC} = 20 400 lb, at a total mass flow of 1000 lb/sec at BPR = 16.1.

Fan Pressure Ratio 1.30–Performance

The performance (thrust and fuel flow) for the 1.30 fan pressure ratio engine is presented in figures B-11 through B-17. These data are for the reference engine with F_{MC} = 22 400 lb at a total mass flow of 1000 lb/sec at BPR = 13.3.

Fan Pressure Ratio 1.35–Performance

The performance (thrust and fuel flow) for the 1.35 fan pressure ratio engine is presented in figures B-18 through B-24. These data are for the reference thrust engine with F_{MC} = 24 000 lb at a total mass flow of 1000 lb/sec at BPR = 11.6.
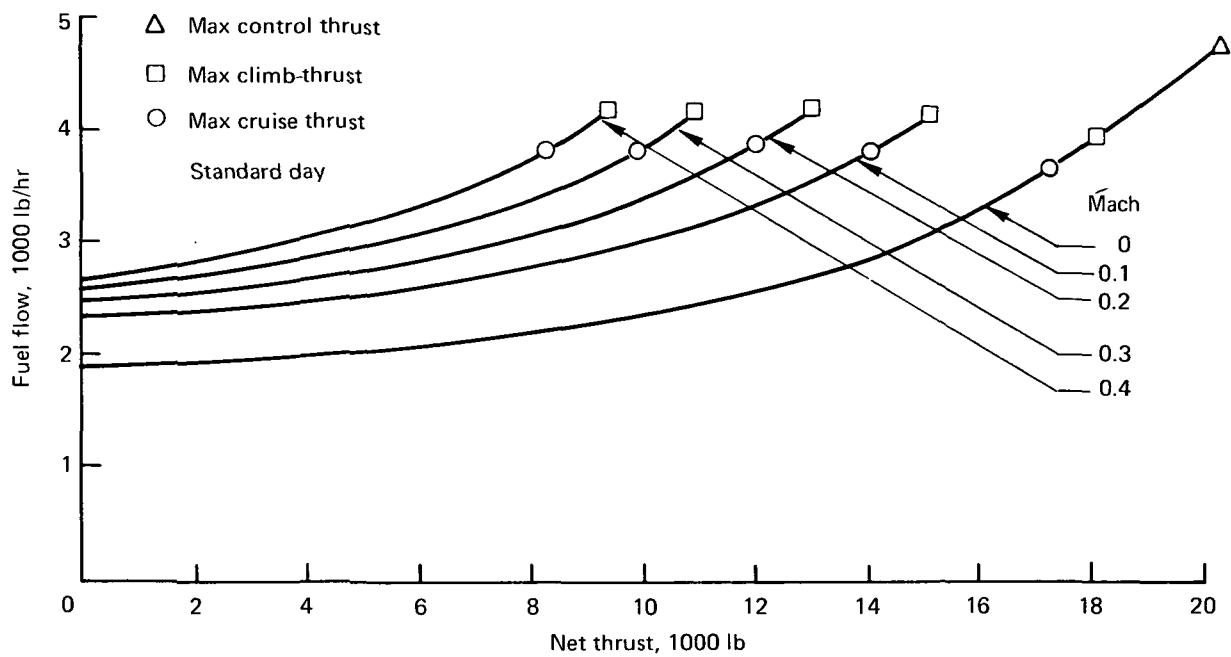
FIGURE B-4.—REMOTE FAN, R_F 1.25, AT SEA LEVEL
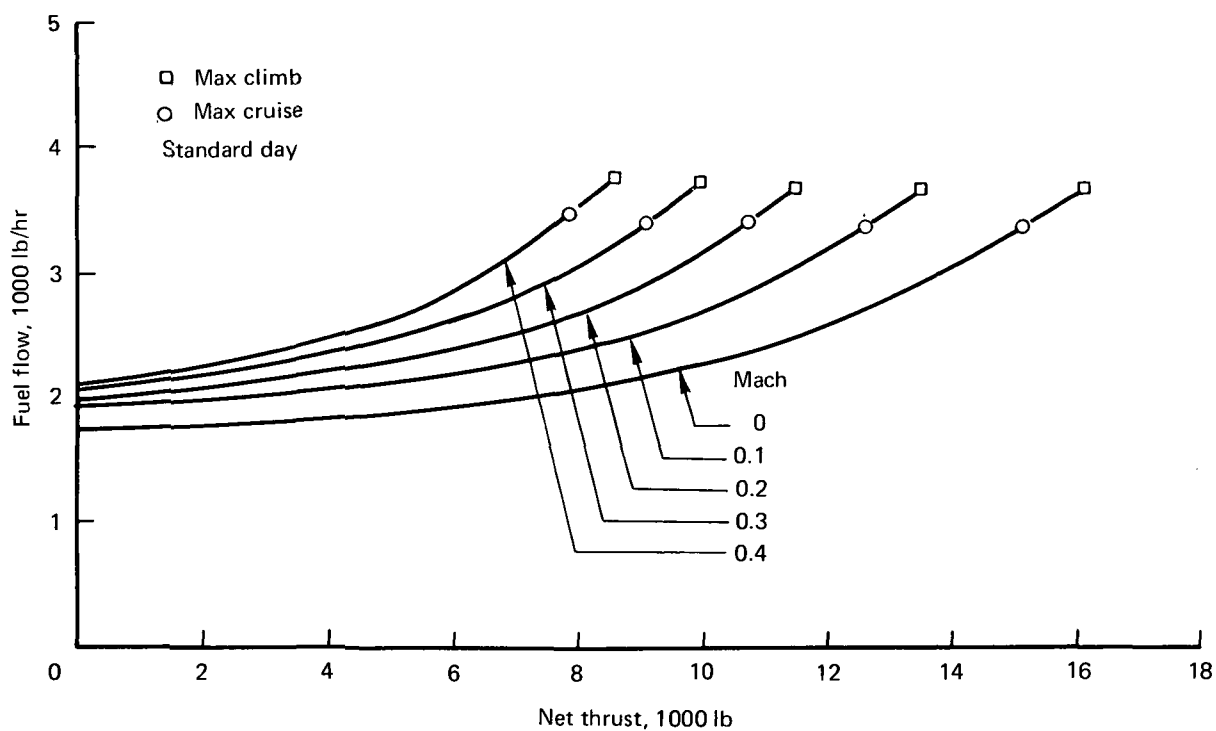
FIGURE B-5.—REMOTE FAN, R_F 1.25, AT 5000 FT
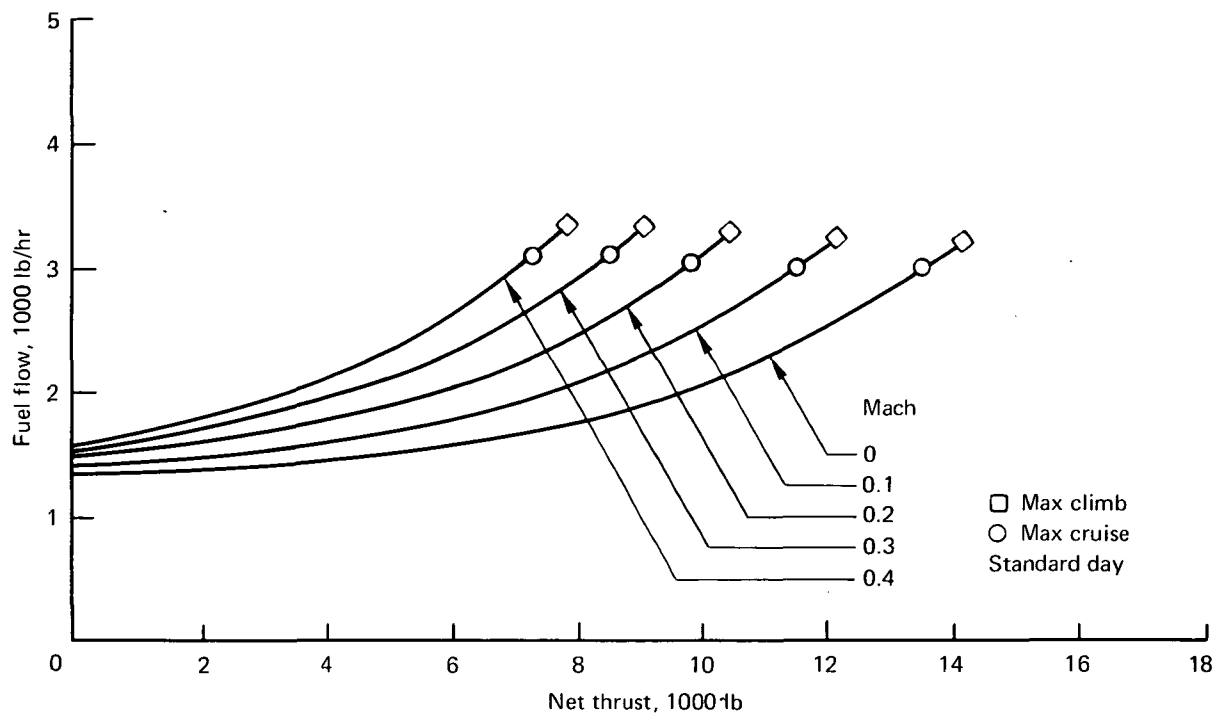
FIGURE B-6.—REMOTE FAN, R_F 1.25, AT 10 000 FT—MACH 0 TO 0.4
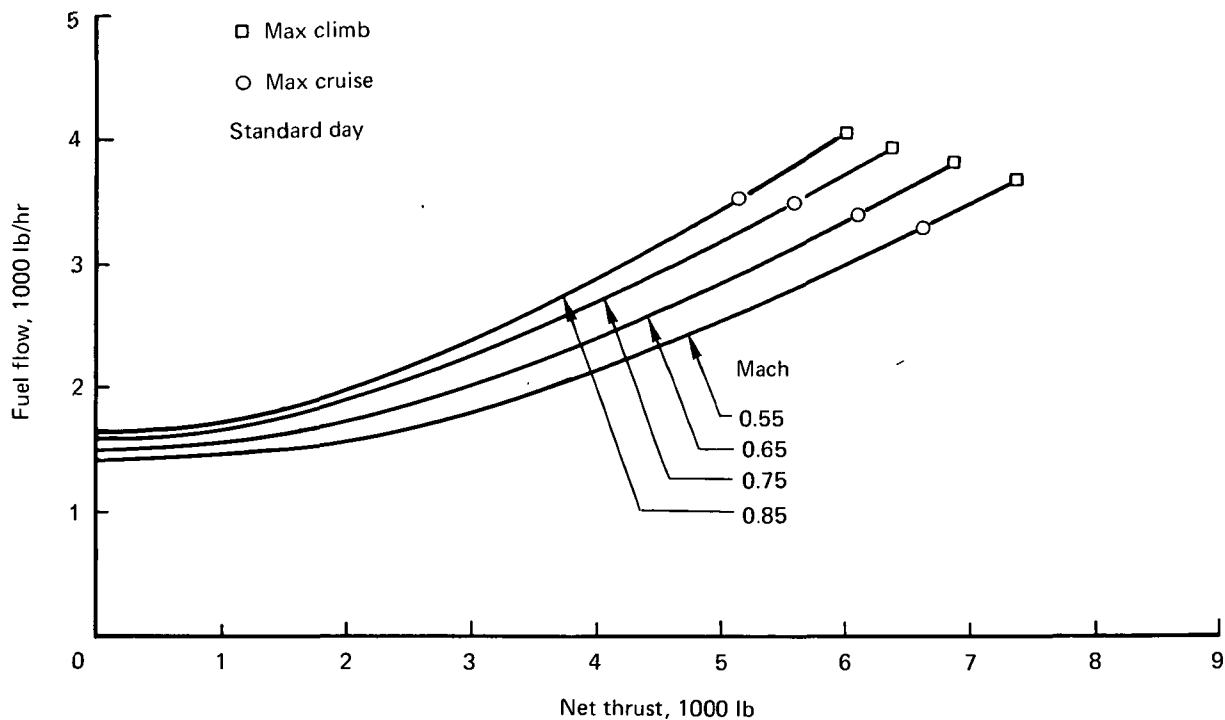

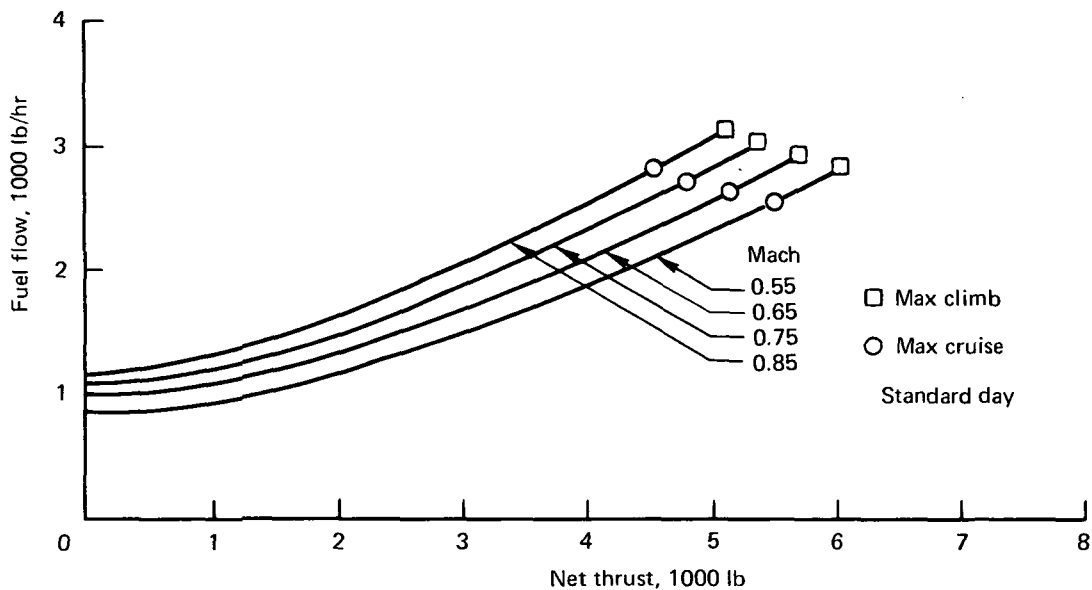FIGURE B-7.—REMOTE FAN, R_F 1.25, AT 10 000 FT—MACH 0.55 TO 0.85
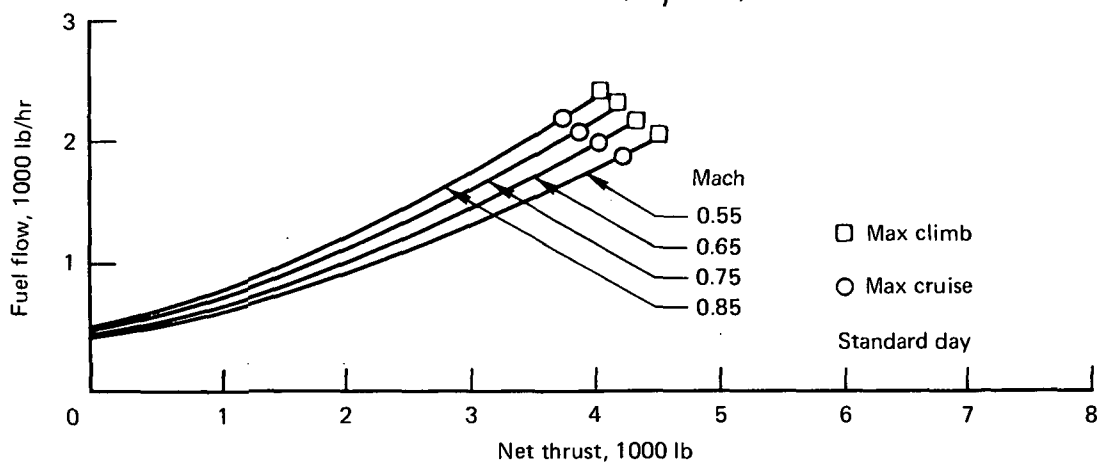
FIGURE B-8.—REMOTE FAN, R_F 1.25, AT 20 000 FT

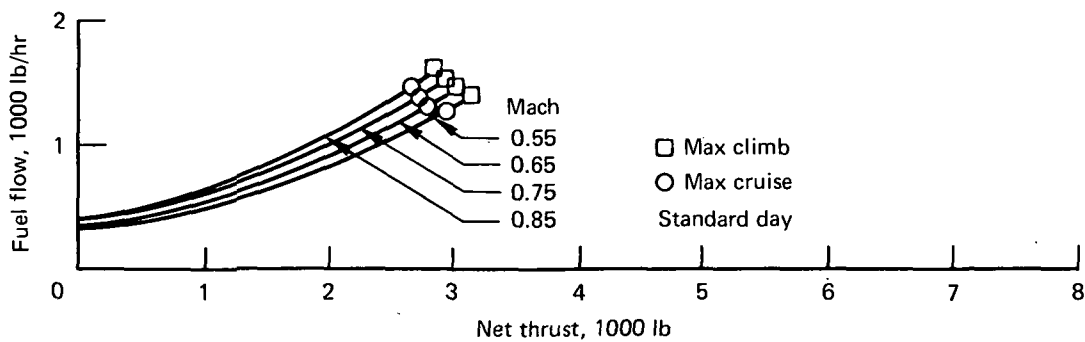FIGURE B-9.—REMOTE FAN, R_F 1.25, AT 30 000 FT
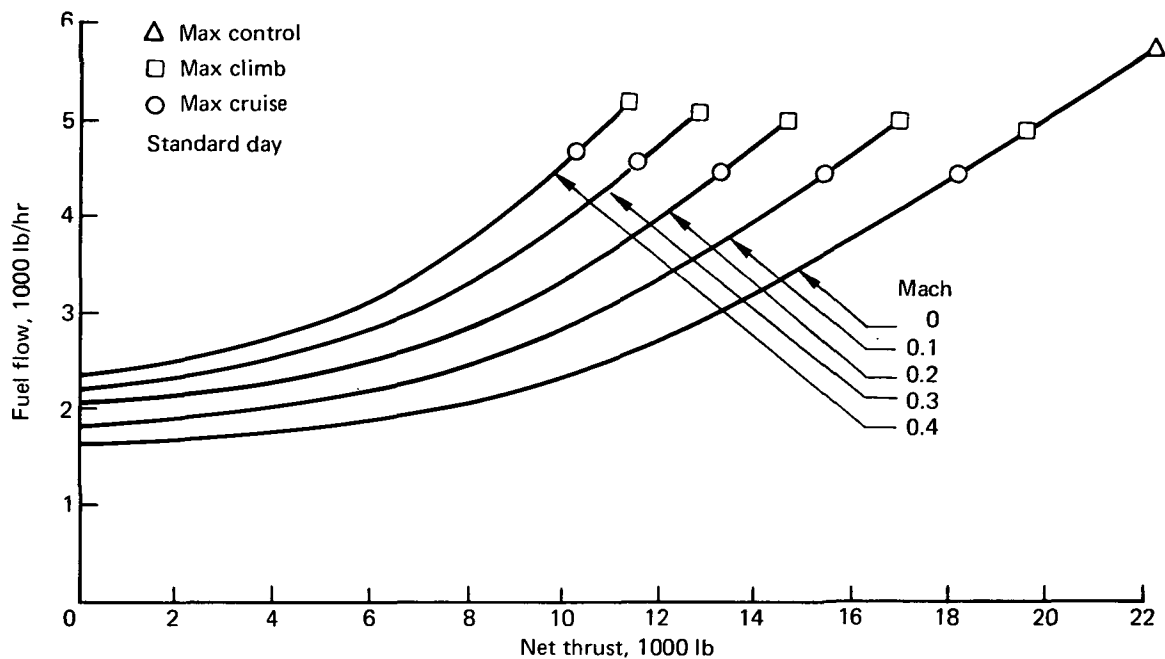

FIGURE B-10.—REMOTE FAN, R_F 1.25, AT 40 000 FT
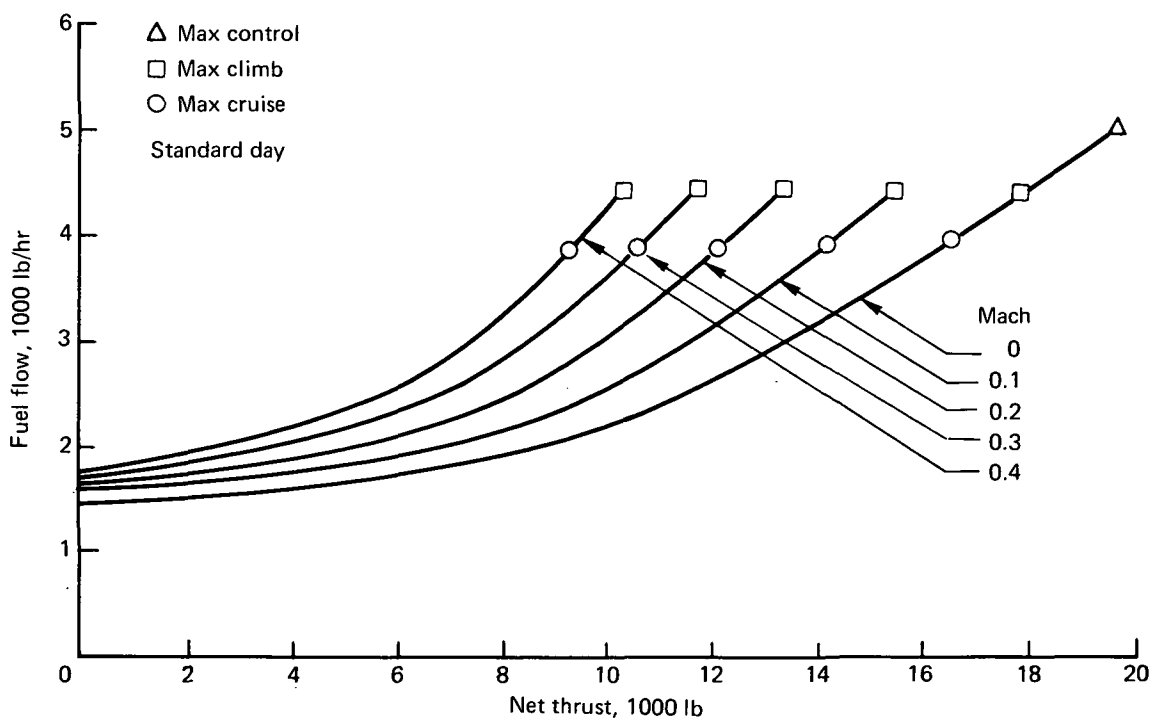
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FIGURE B-11.—REMOTE FAN, R_F 1.3, AT SEA LEVEL
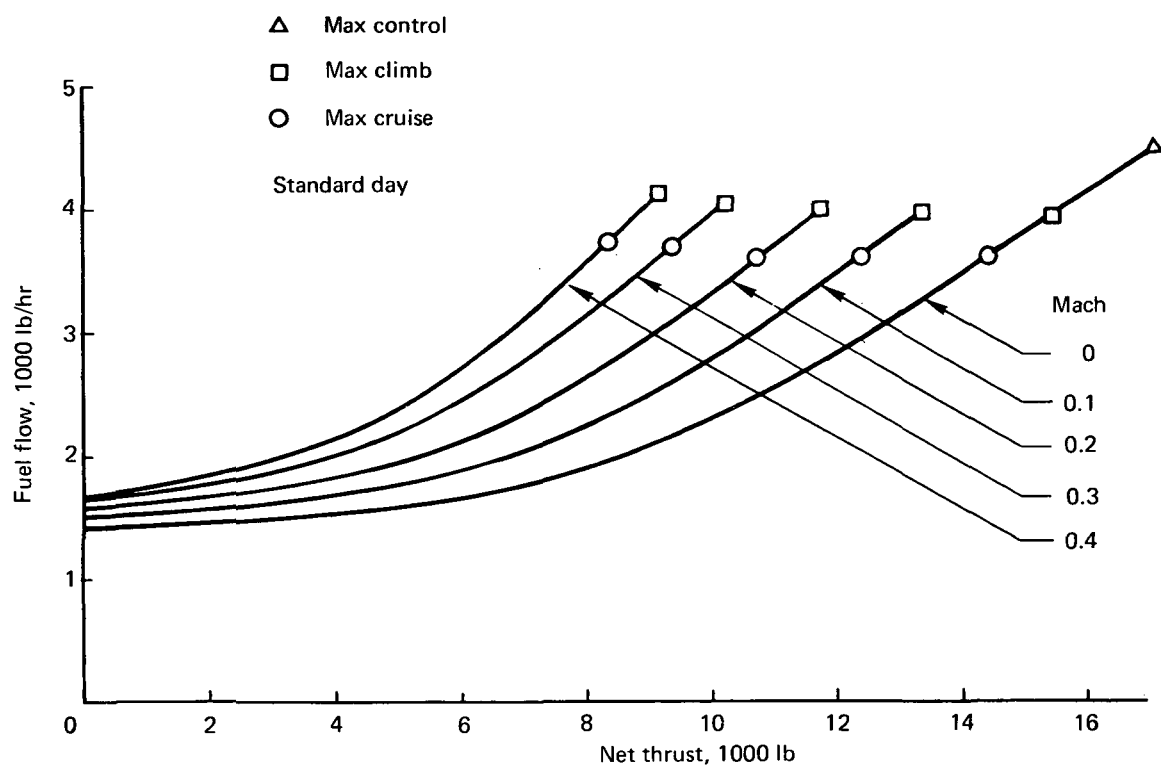

FIGURE B-12.—REMOTE FAN, R_F 1.3, AT 5000 FT

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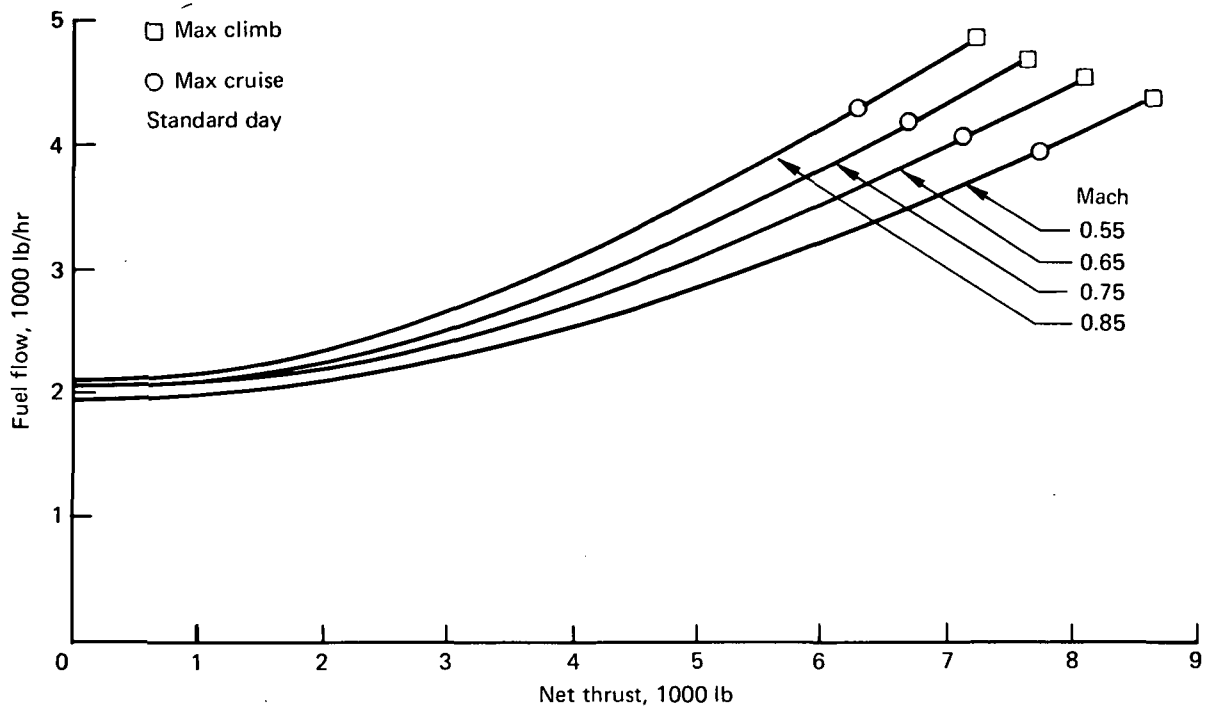FIGURE B-13.—REMOTE FAN, R_F 1.3, AT 10 000 FT—MACH 0 TO 0.4

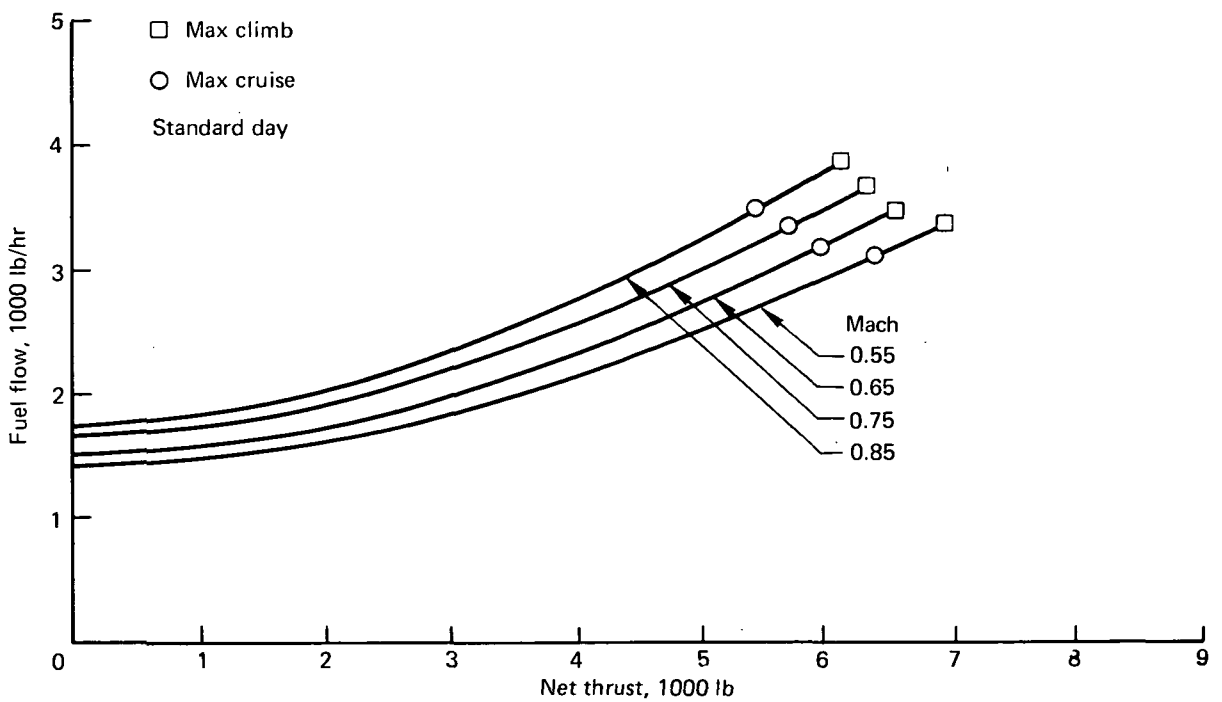FIGURE B-14.—REMOTE FAN, R_F 1.3, AT 10 000 FT—MACH 0.55 TO 0.85

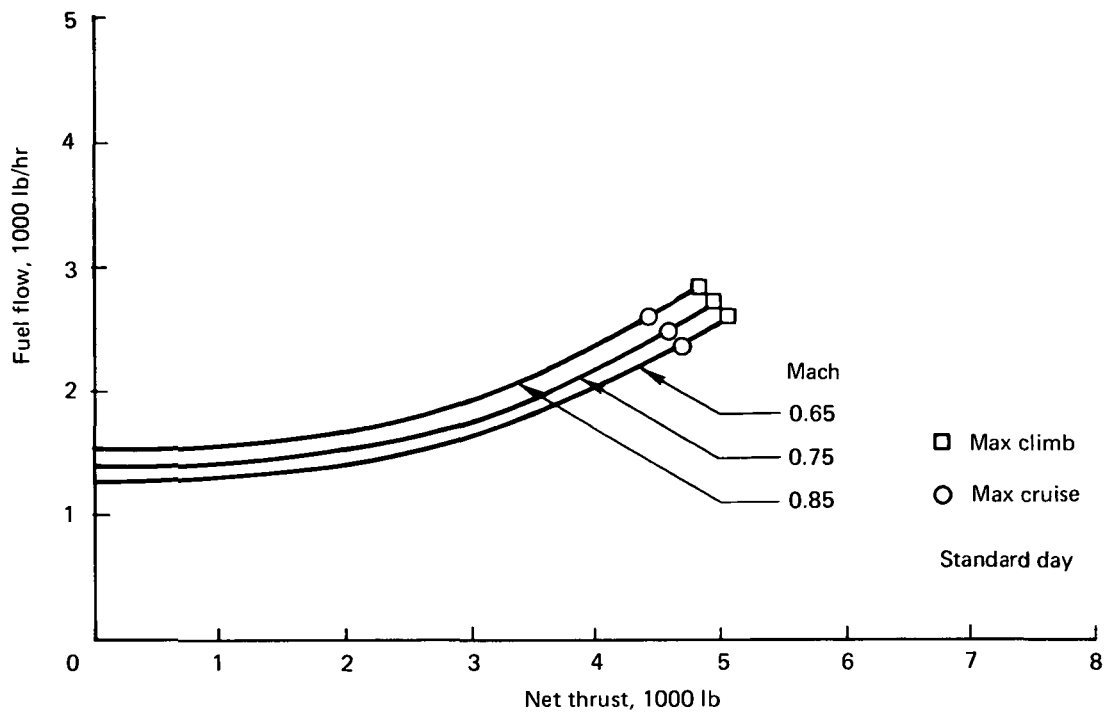FIGURE B-15.—REMOTE FAN, R_F 1.3, AT 20 000 FT

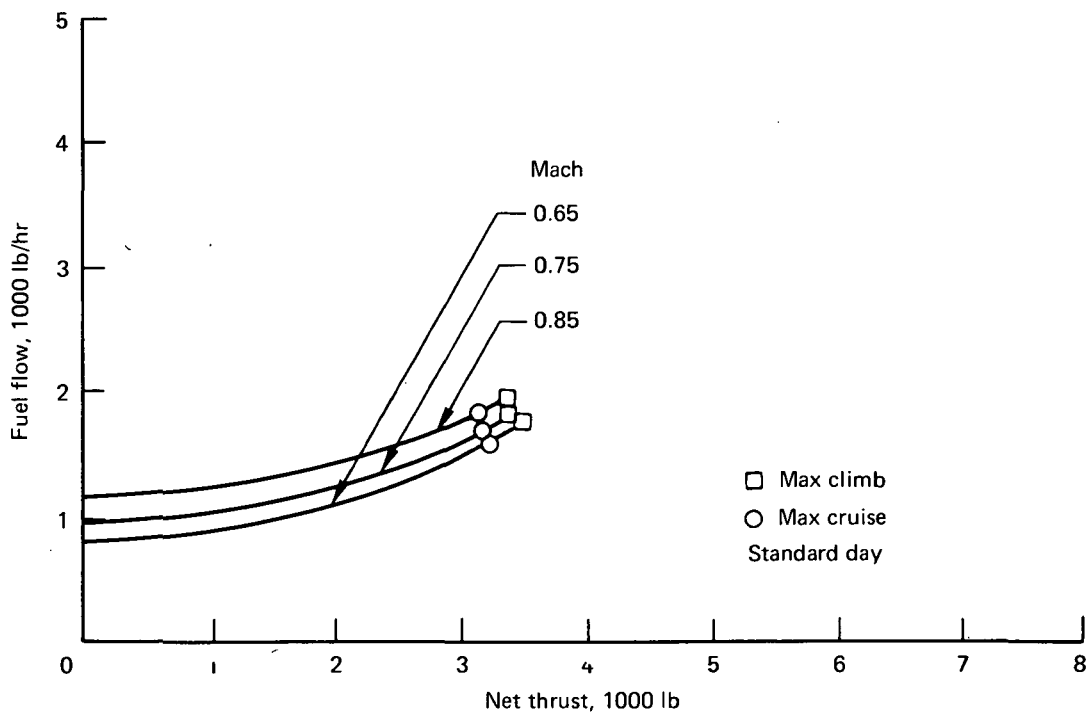*FIGURE B-16.—REMOTE FAN, R_F 1.3, AT 30 000 FT*

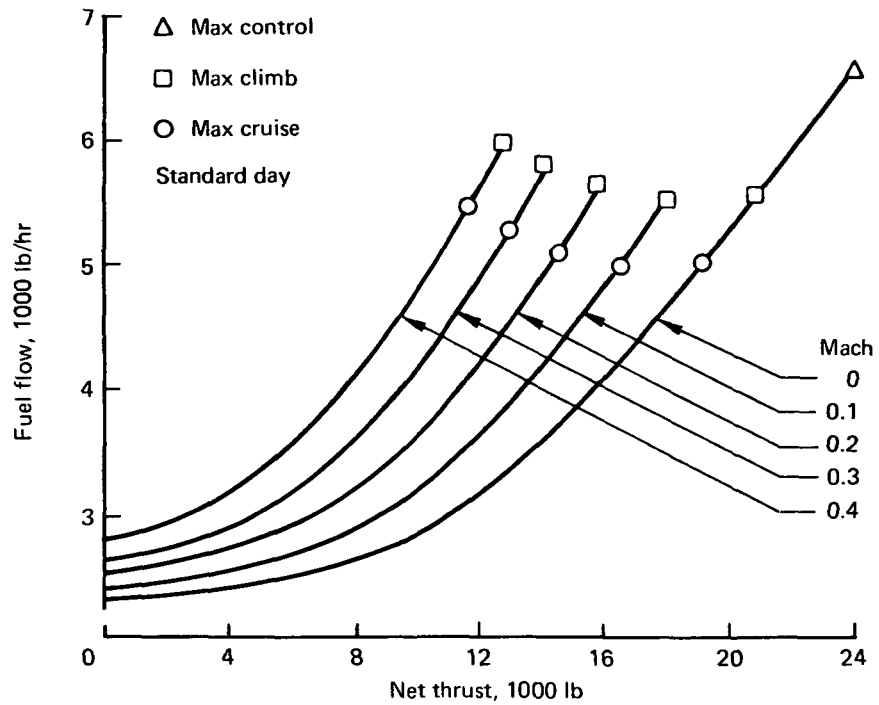
FIGURE B-17.—REMOTE FAN, R_F 1.3, AT 40 000 FT

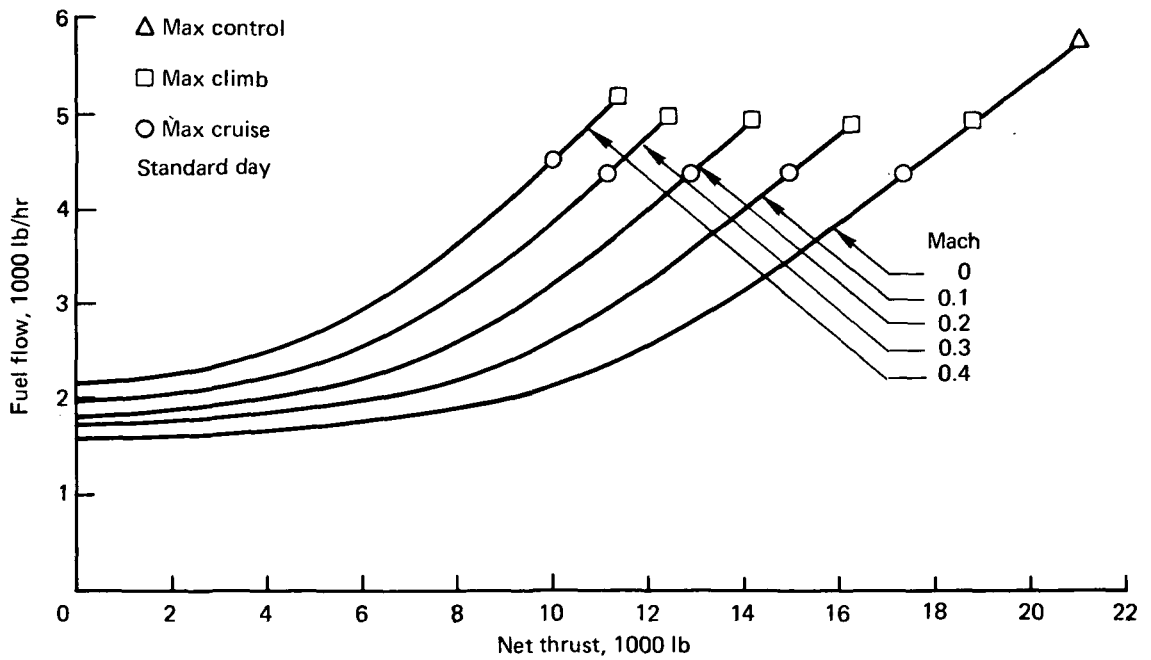FIGURE B-18.—REMOTE FAN, R_F 1.35, AT SEA LEVEL

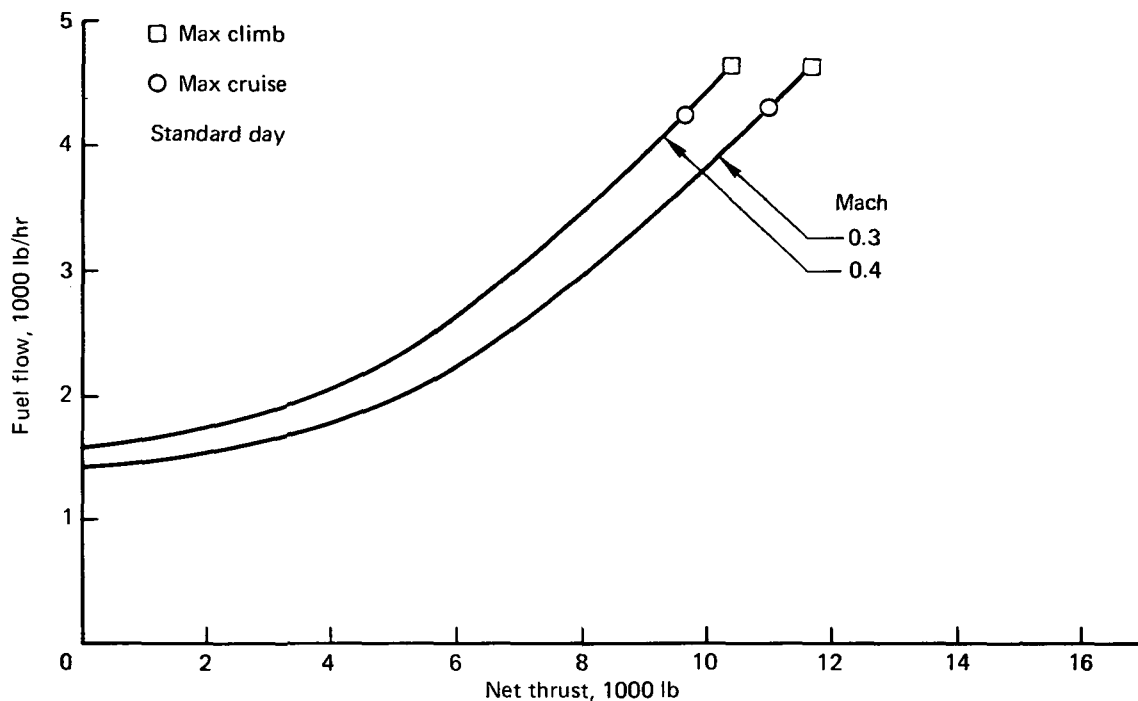FIGURE B-19.—REMOTE FAN, R_F 1.35, AT 5000 FT

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FIGURE B-20.—REMOTE FAN, R$_F$ 1.35, AT 10 000 FT—MACH 0.3 AND 0.4
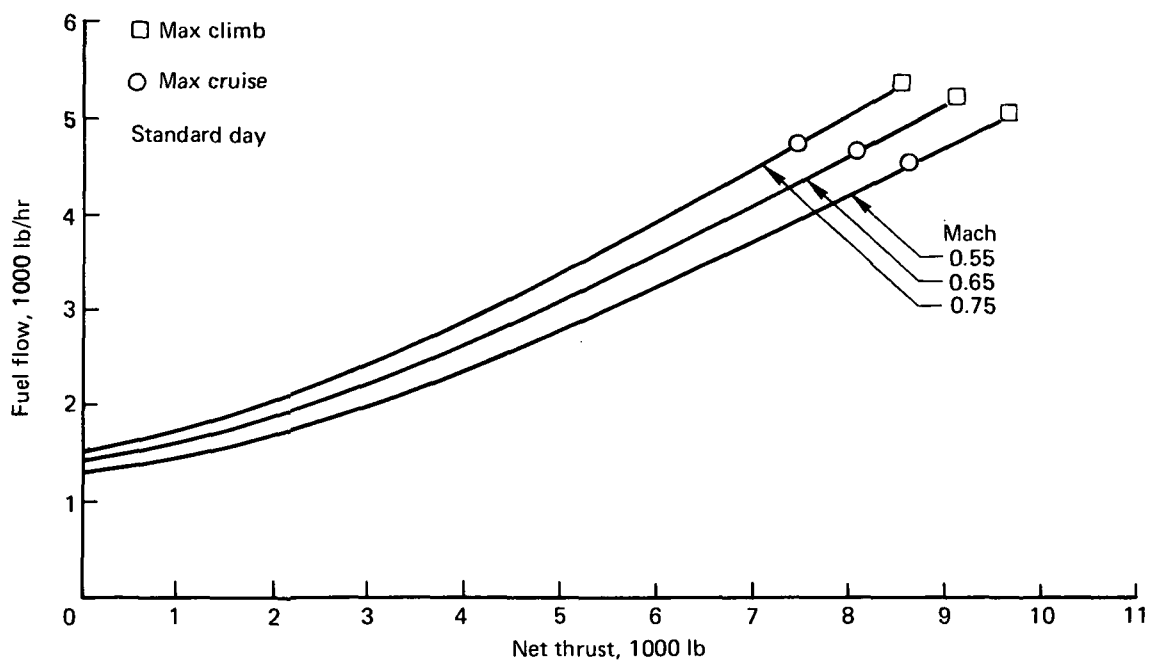

FIGURE B-21.—REMOTE FAN, R$_F$ 1.35, AT 10 000 FT—MACH 0.55 TO 0.75
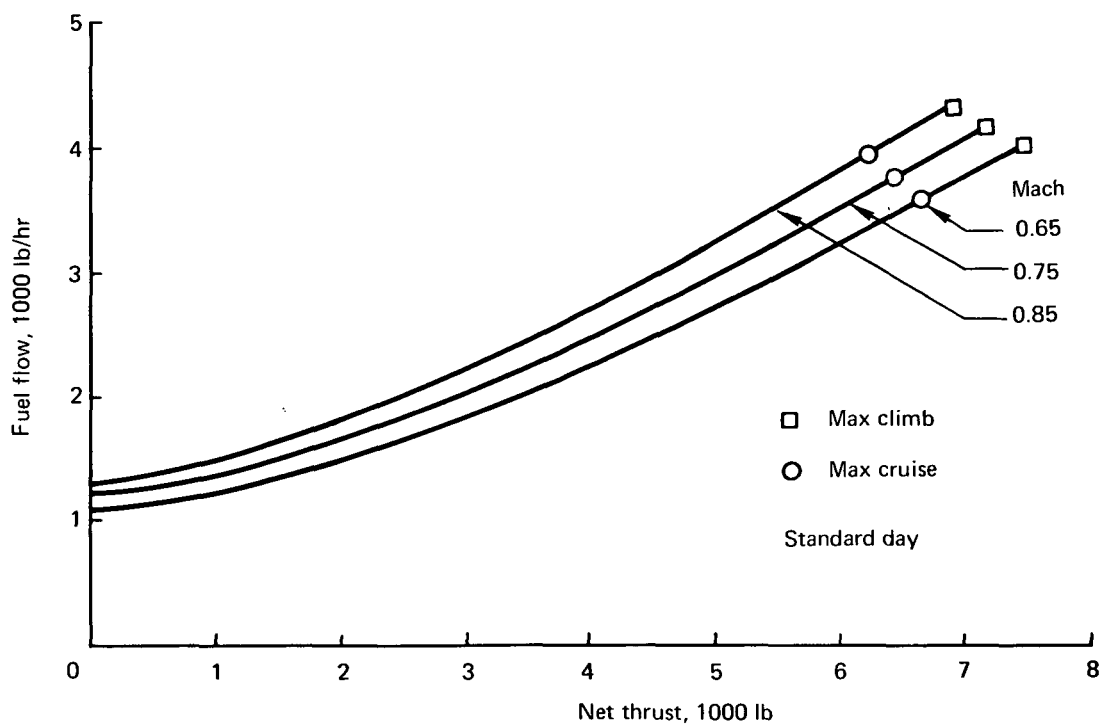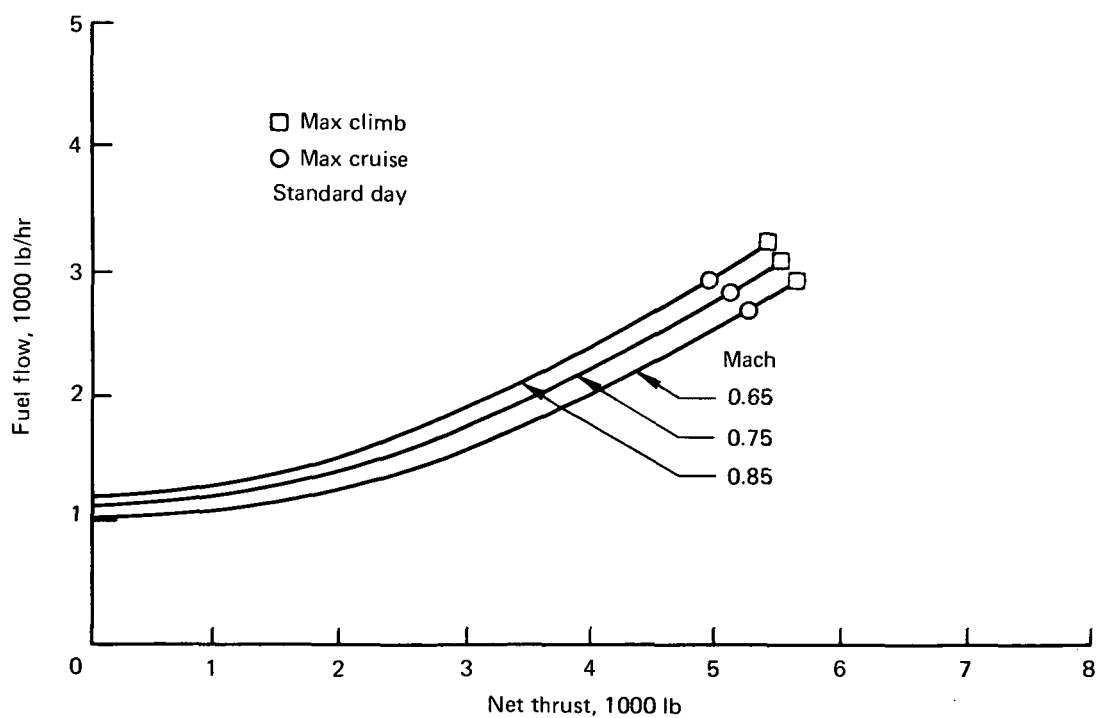
FIGURE B-22.—REMOTE FAN, R_F 1.35, AT 20 000 FT
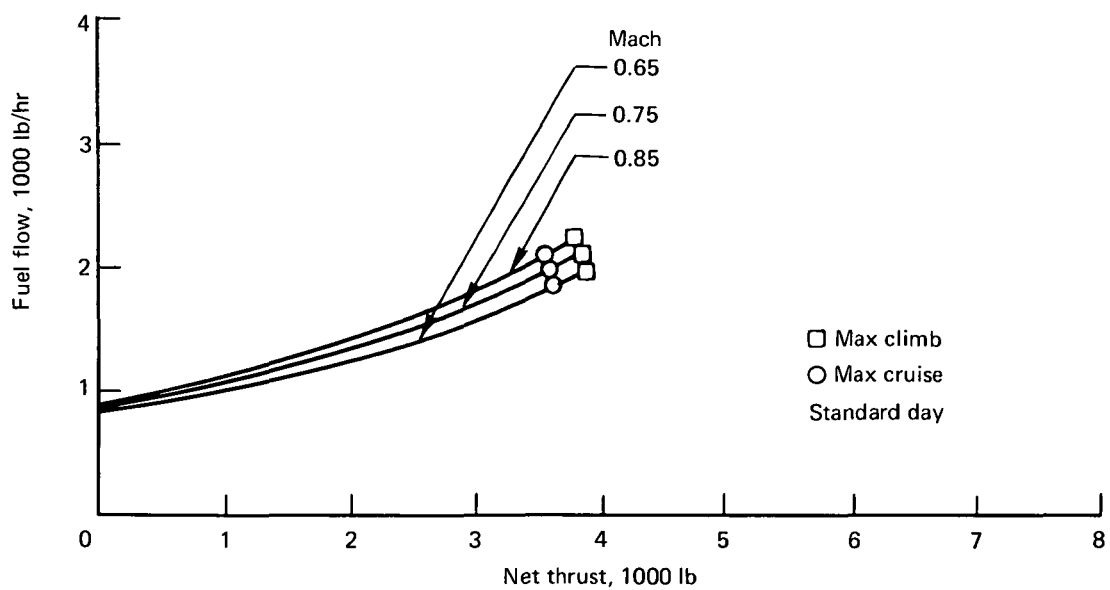

FIGURE B-23.—REMOTE FAN, R_F 1.35, AT 30 000 FT

FIGURE B-24.—REMOTE FAN, R_F 1.35, AT 40 000 FT

APPENDIX C

CONFIGURATION SELECTION

Initially, configurations with six, seven, or eight fan/gas generator combinations were conceived to accomplish the baseline mission. A six-engine airplane was chosen as the best match between the VTO and cruise requirements. A series of six-engine configurations were studied as part of the selection process. The initial arrangement, figure C-1, had four fans on the fuselage and two at the wingtips. The forward fuselage fans were stowed in the fuselage during conventional flight. The alternative configurations, shown in figures C-2 to C-6, represent attempts to avoid this feature.

Figure C-2 shows the nose fans in a fixed position as the only change from the design in figure C-1. This modification results in an L/D reduction of about 5-1/2% and an empty weight increase of about 1000 lb.

The configuration in figure C-3 has two fans at each wingtip and two at the tail. The cruise fan pitch moment arms are three times longer than the wing fan pitch moment arms. The wing fan overhang was limited by the twisting moment on the wing. The six fans are all interconnected and the VTOL F/W required is 1.37. This increase over F/W = 1.32 is the result of the unequal moment arms, which result in throttled operation of the cruise engines during V/STOL operation. An empty weight increase due to the wing structure and engine size is estimated to be in excess of 5000 lb.

Another concept with all six fans on the wing is shown in figure C-4. The rear fans are the cruise fans and rotate from vertical to horizontal. In addition to high inertias, the wing must withstand the same twisting moment as the one in figure C-3 in the event of failure of one of the forward or aft fans. A weight penalty also in excess of 5000 lb is involved.

Another design without folding fans, figure C-5, has the forward fans contained in a strake forward of the wing and the mid fans mounted between the wing spars, in line with the forward and cruise fans. All six fans are used for roll control. This arrangement was an attempt to correct the faults of the design shown in figure C-4. However, the longitudinal placement of the fans, the wing structure, and the fan fairing have given this airplane an empty weight increase of 4200 lb.

A tandem-wing airplane, shown as figure C-6, represents another attempt to avoid folding fans. The forward fans are housed in the forward wing and determined the wing size. The aft wing is sized to provide the desired stability. The resulting wing loading is too low and the empty weight is 2100 lb greater than the baseline in figure C-1.

The design with folding fans, although more complex from a machinery standpoint, is cleaner and lighter than those fixed designs.
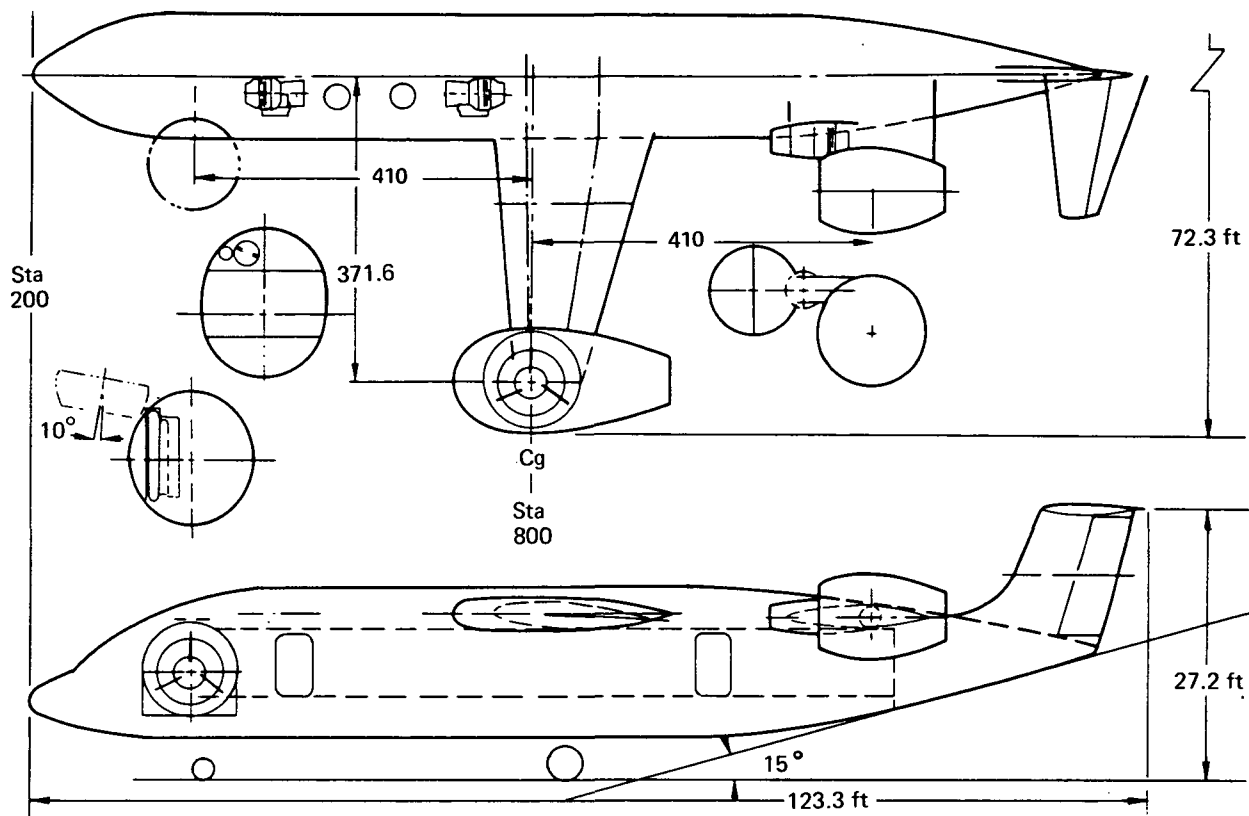
FIGURE C-1.—MODEL 984-136 (SIX-FAN/SIX-ENGINE)
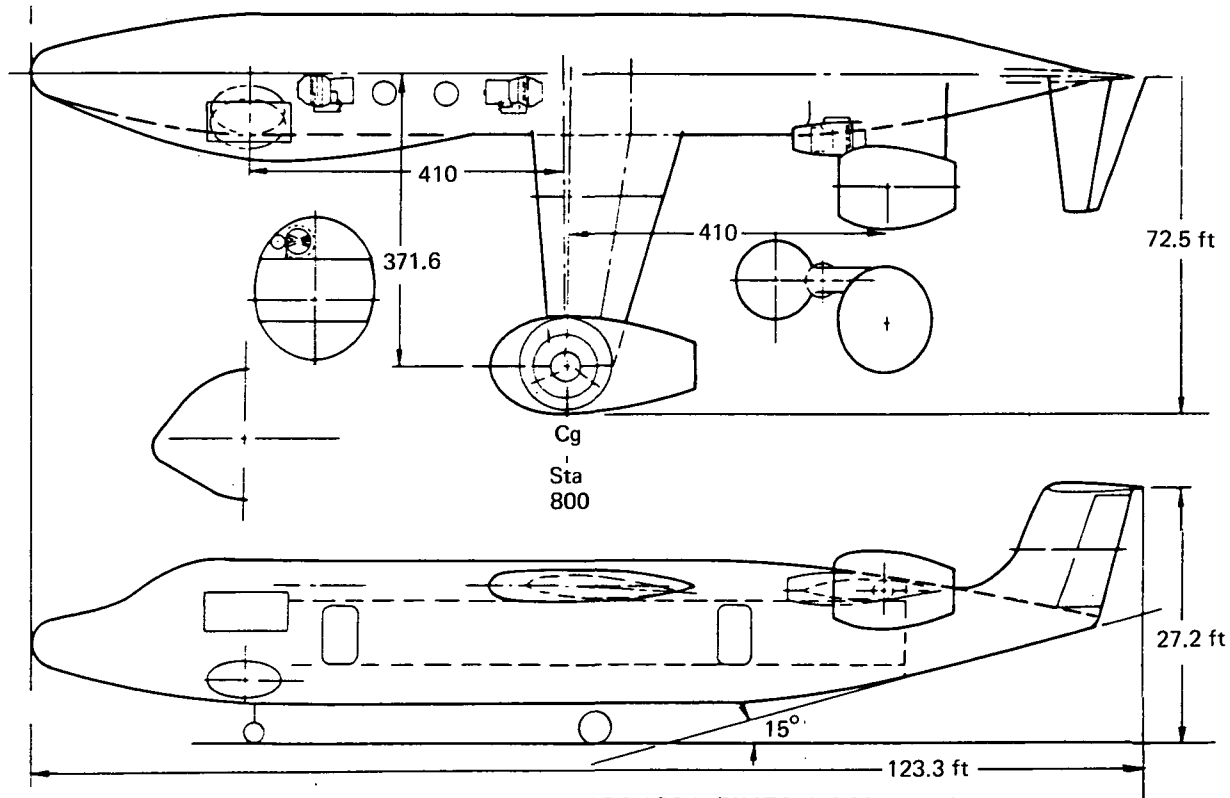

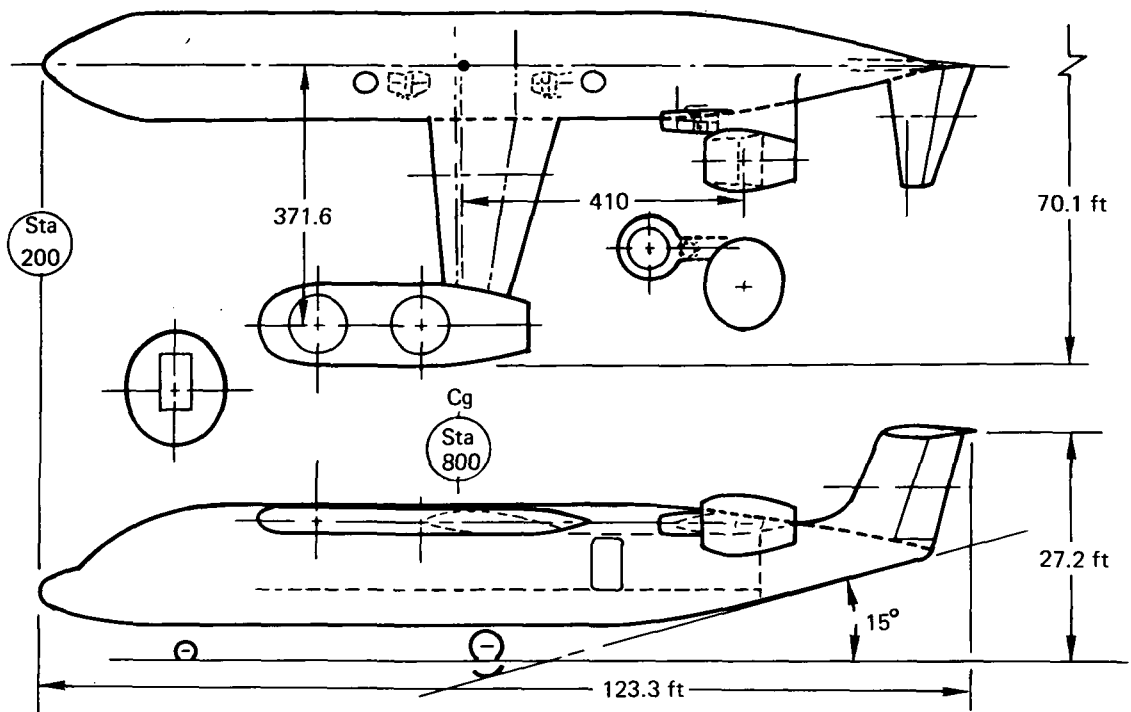FIGURE C-2.—MODEL 984-136A FIXED NOSE FANS
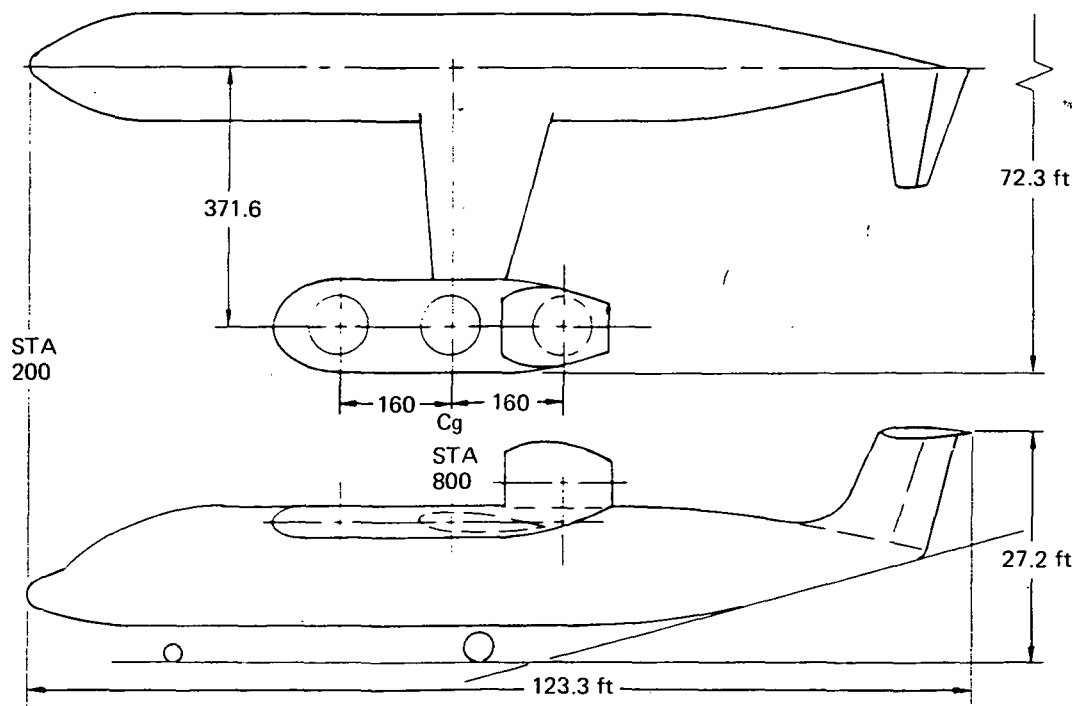
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FIGURE C-3.—MODEL 984-136B LIFT FANS AT WING TIP
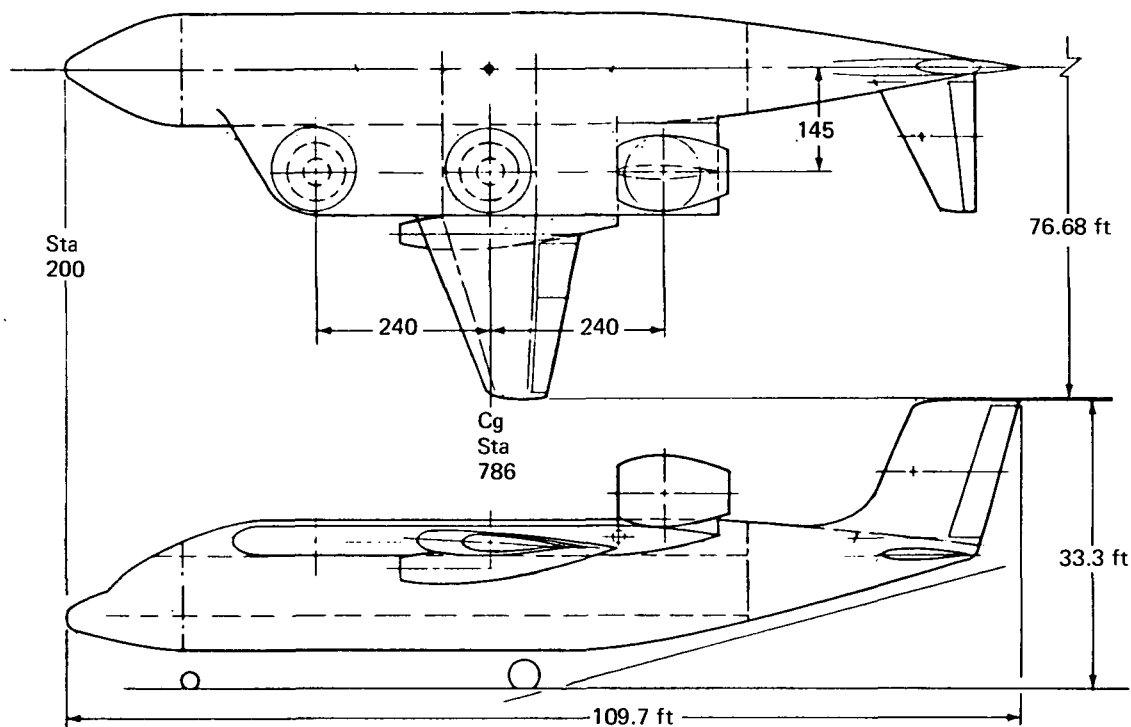

FIGURE C-4.—MODEL 984-136C FANS ON WING

FIGURE C-5.—MODEL 984-143 FANS ON FUSELAGE

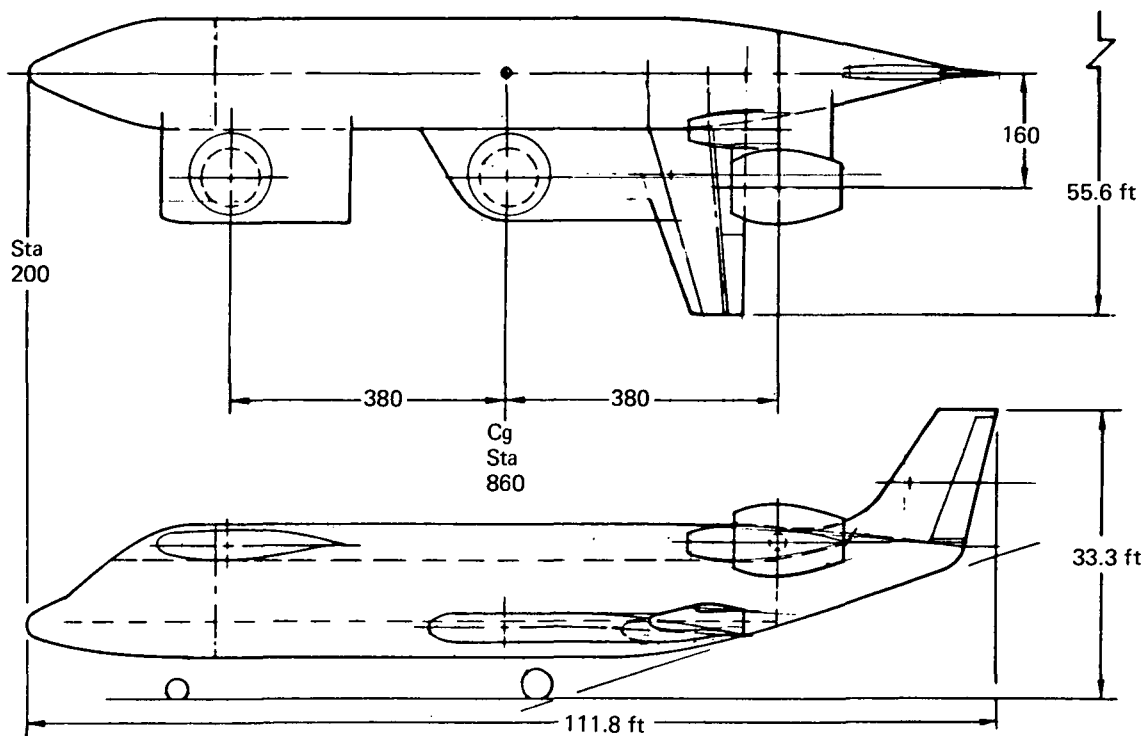FIGURE C-6.—MODEL 984-140 TANDEM WING

APPENDIX D

WEIGHT AND BALANCE

The installation of thrust greater than the airplane weight dominated the weight of these airplanes. The basic aircraft that resulted from this study are the six-engine baseline airplane (984-139), the four-engine, "no fan failure" 200-nmi airplane (984-144), and the six-engine 150-passenger airplane (984-145). Isometric views are shown in the summary (fig. 2). Weight and balance statements for these airplanes are presented in table D1.

The aircraft are all balanced at 35% of the mean aerodynamic chord at the maximum VTO gross weight. The balance and loadability were analyzed to achieve a wing-body-engine relationship yielding center-of-gravity limits that minimize static pitch trim requirements of the engines during hover and yet maintain a flexible passenger loading capability. The optimum engine lift center was determined at 35% of the MAC; therefore, the engines were arranged so that the lift center was at 35% of the MAC and the wing was placed on the body so that the resulting center-of-gravity limits would vary as little as possible from 35% of the MAC. The following assumptions were used.

a) The balance and loadability was predicated upon the hover case and not the conventional flight case. This condition was considered the most severe.

b) The main landing gear was shifted with the wing so that it stayed at a constant distance behind the wing quarter chord.

c) A ± 2% MAC tolerance was assumed on the probable passenger-plus-fuel loading conditions to determine the fore-and-aft center-of-gravity limits. This assumption was the result of such considerations as in-flight movement of passengers, shifts in fuel cg due to changes in aircraft attitude, and operator variances.

d) For the six-lift-fan configurations, the forward body lift fans were fixed at their location between the flight deck and the passenger cabin. Consequently, the aft cruise lift fans were moved as the wing was shifted so that the VTO lift center was at 35% of the MAC (the wing lift fans were located at 35% of the MAC and remained there as the wing was shifted).

e) For the four-lift-fan configuration, the lift and cruise-lift fans were balanced about the 35% MAC and moved with the wing to maintain the VTO lift center at 35% MAC.

TABLE D1.— V/STOL ADVANCED TECHNOLOGY WEIGHT AND BALANCE

Model 984	Six engines, 100 passengers 984-139		Four engines, 100 passengers 984-144		Six engines, 150 passengers 984-145	
	Weight (lb)	Cg (in.)	Weight (lb)	Cg (in.)	Weight (lb)	Cg (in.)
Wing	7 000	651	5 750	808	11 430	955
Horizontal tail	990	1 308	860	1 500	1 300	1 769
Vertical tail	1 120	1 272	890	1 435	1 600	1 717
Body	15 950	623	13 800	807	24 750	907
Main landing gear	4 450	640	3 880	842	6 840	987
Nose landing gear	490	230	390	363	760	503
Nacelle and strut	12 160	704	10 900	783	18 810	991
Air stairs	400	327	400	475	400	573
Total structure	(42 560)	(676)	(36 870)	(828)	(65 890)	(975)
Engine	21 700	635	19 360	740	37 430	918
Engine accessories	810	635	670	740	1 080	918
Engine controls	260	540	140	708	300	828
Starting system	400	635	280	767	400	918
Fuel system	720	634	580	795	950	945
Thrust reverser						
Duct system	6 780	585	3 470	835	11 270	902
Total propulsion system	(30 670)	(623)	(24 500)	(755)	(51 430)	(914)
Instruments	990	302	970	480	1 060	545
Surface controls	3 620	726	3 280	880	4 770	1 048
Hydraulics	1 140	806	1 060	960	1 420	1 142
Pneumatics	350	635	310	786	490	918
Electrical	1 030	470	1 030	640	1 030	750
Electronics	1 200	289	1 200	470	1 200	538
Flight provisions	760	134	790	327	890	357
Passenger accommodations	6 400	635	6 550	750	9 500	914
Cargo handling	540	325	270	525	810	609
Emergency equipment	350	497	300	668	500	781
Air conditioning	1 190	672	1 190	832	1 440	985
Anti-icing	650	635	610	832	820	985
Auxiliary power unit	780	1 125	780	1 005	900	1 510
Total fixed equipment	(19 000)	(605)	(18 340)	(744)	(24 830)	(906)
Exterior paint	100	672	100	835	100	985
Options						
Manufacturer's empty weight						
Standard and operational items	2 420	413	1 630	620	3 250	720
Operating empty weight	94 750	638	81 440	783	145 500	936
Maximum zero fuel weight						
Maximum VTO weight	126 300		107 900		192 500	

f) For the long-range, single-aisle, passenger cabin configurations, passenger loadability was assumed to follow the pattern of window seats filled first, then the aisle seats, and finally the remaining seats.

g) For the double-aisle passenger cabin, the loading pattern was window seats first, then window aisle seats, and finally the center aisle seats. Both front-to-rear and rear-to-front loading were used to determine the range of probable passenger loading.

h) The loading diagrams for the three sized configurations are shown in figures D-1, D-2, and D-3.

INERTIA PROPERTIES

Inertias were developed for each configuration by analysis of mass properties about three axes at the maximum vertical takeoff weights. The incorporation of a T-tail empennage was the major contributor to the nose-down inclination of the principal axis.

Table D2 presents the values of the inertias for these aircraft.

METHODOLOGY

Several techniques for deriving configuration empty weights were available and virtually all of them were utilized in one aspect or another. In the areas of structures, systems, and fixed equipment the basic estimating tool is a Boeing preliminary design weight analysis method for V/STOL aircraft. The method provides an initial weight estimate based on variables descriptive of aircraft geometry, payload requirements, guideline criteria, propulsion characteristics, and performance. The initial estimate was modified where sufficient details were available to allow a more complex weight estimation to be performed. In addition, the study guideline criteria provided a source of weight values for some of the systems and equipment components of the configurations.

Each configuration in this study is powered by a General Electric remote lift-fan propulsion system. The system weights for engines, louvers, and duct components were developed for the General Electric data.
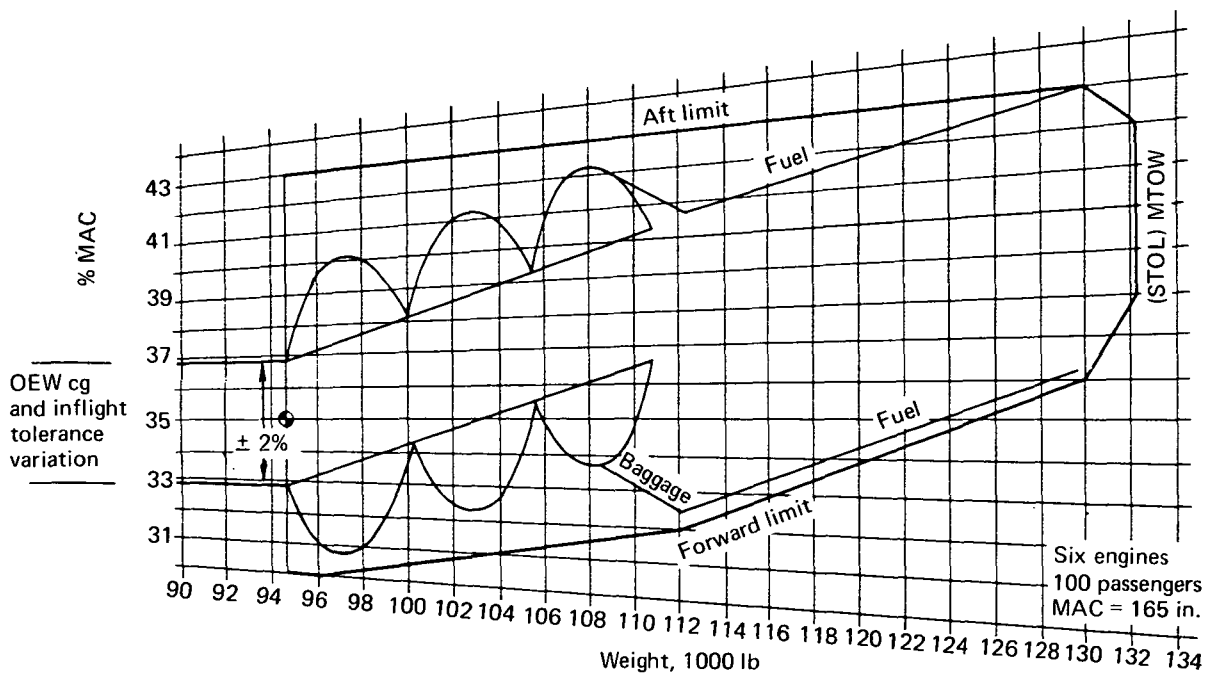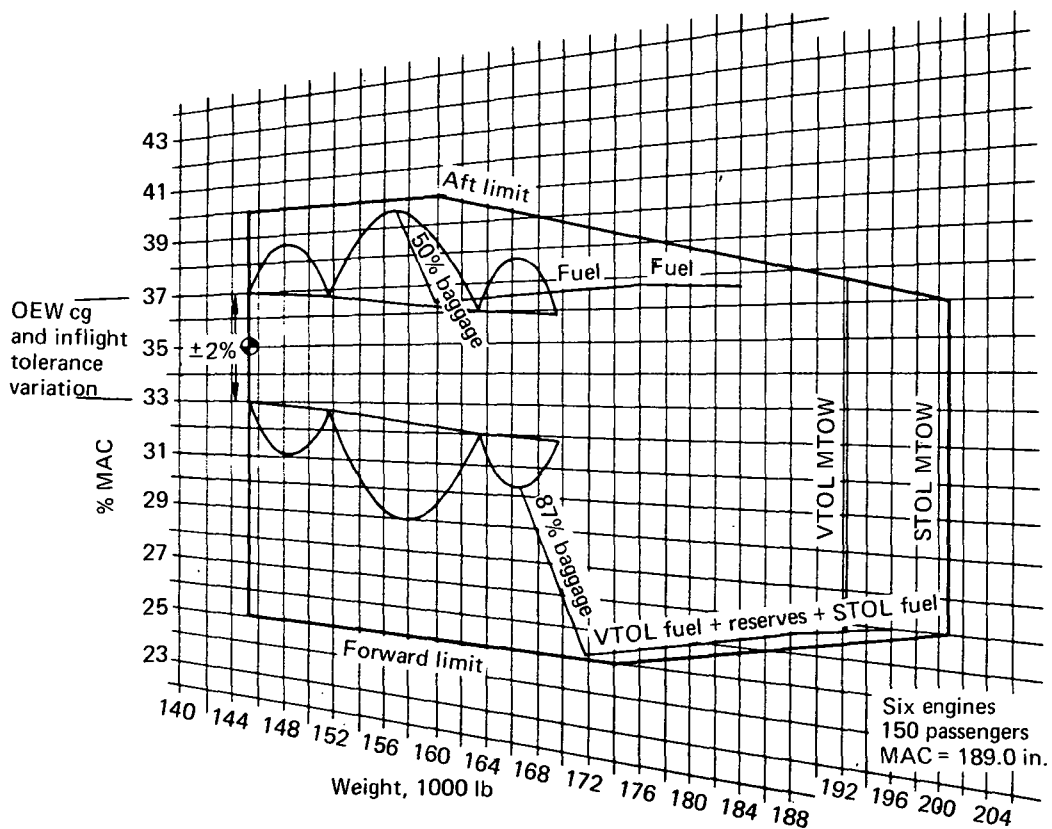
FIGURE D-1.—MODEL 984-139
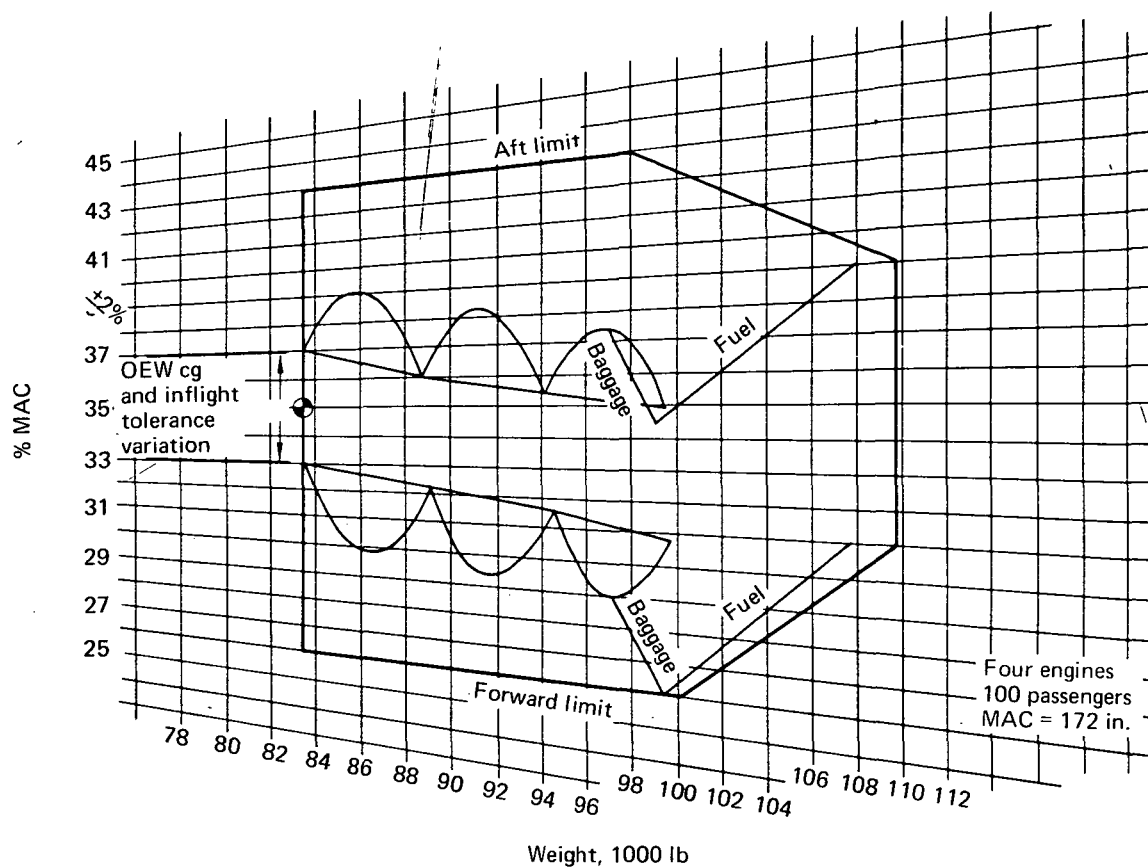

FIGURE D-2.—MODEL 984-145

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FIGURE D-3.—MODEL 984-144

TABLE D 2.—INERTIAS

Airplane	Roll	Pitch	Yaw	(XZ) product
984-139—six engines, 100 passengers	0.53	2.31	2.72	0.075
984-145—six engines, 150 passengers	0.44	1.19	1.51	0.050
984-144—four engines, 100 passenger	1.05	4.4	5.18	0.14

Inertia—slug ft^2 x 10^{-6}
at maximum VTO weight